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LUNAR MISSION SYSTEMS STUDIES

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SPACE and INFORMATION SYST[redacted]MS DIVISION

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1.0 SUMMARY

This material is presented to substantiate the feasibility of accomplishing a direct lunar landing mission with an Apollo command module by utilizing cryogenic propulsion systems on the lunar landing and launch stages. All systems or components have off-the-shelf availability, or are so programmed for the time period under consideration. The ground rules employed specifically directed that the above conditions be met in order that the primary mission objectives of NASA could be realized at lowest cost, highest reliability, and earliest launch date; and alternately to offer a system with growth potential.

Current design criteria impose a net injected weight limit for the Saturn C-5 launch vehicle of 90,000 pounds. The weight of the vehicle considered in this study is less than this value, yet includes conservative estimates wherein time and data were insufficient for thorough analysis. Additionally, this configuration reflects a three-man, fourteen-day mission.

The landing stage of the subject vehicle of this study would make an excellent candidate for the NASA Lunar Logistic Vehicle. The adaptation of the lunar landing stage to the Lunar Logistic Vehicle is shown in S&ID Report SID 62-1466, "Lunar Logistic Vehicle Study."

Areas requiring further examination do exist, but appear to be analytically conquerable. In order of relative importance, these are:

1. Application techniques for super insulation.
2. Detailed propellant system design.

The results of this study show that the mission can be accomplished

with the vehicle considered and with reasonable conservatism in the system weight and vehicle performance.

2.0 INTRODUCTION

This report summarizes the results of an intensive three-month study conducted as independent research and development by North American Aviation, Inc., Space and Information Systems Division, of an advanced, second generation Apollo spacecraft configuration capable of a direct landing mission to the Moon using the C-5 launch vehicle system for injection to escape. This objective can be accomplished by the use of redundant microminiaturized avionics systems equipment and a more efficient low-density heat shield material to reduce payload weight and by the use of a return propulsion system utilizing liquid oxygen and liquid hydrogen as propellants and an advanced version of the Pratt & Whitney RL-10 engine. This approach permits the direct landing of three men on the surface of the Moon for extended scientific surface activities. In addition, the requirement for the use of the rendezvous technique in either Earth or lunar orbit is eliminated along with the requirement for dual spacecraft or launch vehicle systems and the attendant reliability and crew safety hazards associated with the rendezvous approaches to lunar landing operations. Use of the lunar orbital rendezvous technique significantly restricts the available lunar latitudes for landing sites as well as the stay time on the lunar surface. Successful rendezvous with the orbiting spacecraft is mandatory for safe return to the Earth. Studies conducted by Space and Information Systems Division under study contract to the National Aeronautics and Space Administration have indicated a significant increase in the complexity of the required spacecraft crew tasks in the lunar orbital rendezvous mission as compared to the direct flight technique. These complexities, as well as the effects of increased hardware complexity, have made it desirable to explore carefully the direct mission alternative to the lunar landing.

Studies conducted during the past year have clearly indicated that the full three-man, fourteen-day Apollo mission can be accomplished as indicated above with single Saturn C-5 launch. Such studies were previously summarized in Interim Progress Report, SID 62-1189. Additional details on propulsion aspects of such a vehicle were presented in Report SID 62-1189-1. Still other studies of what might be termed more radical spacecraft concepts, including the use of the propellant combination fluorine/hydrogen, were reported in SID 62-1190.

In response to a NASA RFP, Report SID 62-1050 was produced for the proposed study of the lunar logistic vehicle concept for support of lunar operation. The approach taken in this study proposal was not to design another new independent spacecraft but rather to consider the logistic vehicle (LLV) in context with the manned lunar landing system. It was readily apparent that the two systems could be made entirely compatible without duplication by careful attention and merging of the two mission requirements (this is in contrast to the present trend wherein the LOR/LEM manned mission hardware is independent of the logistic mission hardware). These studies have shown that the three-men, direct fourteen-day mission spacecraft, employing high-energy propellants, is then directly adaptable for the lunar logistic mission. The first, or landing, stage for the manned mission is adaptable with minor modifications for use as the LLV. This is the philosophy employed in the studies reflected in this report and in Report SID 62-

1466 covering the lunar logistic vehicle.

3.0 CONCLUSIONS & RECOMMENDATIONS

1. The full three-man, fourteen-day Apollo mission is feasible within the present program time span employing single C-5 launch (no rendezvous operation required) by employing high-energy oxygen/hydrogen propellants.

2. The advanced version of the Pratt & Whitney RL-10 engine is suitable for use in the subject manned vehicle and will have demonstrated sufficient reliability to meet the manned requirement. The engine will be capable of being throttled to 10% of maximum thrust for the lunar touch-down maneuver.

3. The Apollo spacecraft will require the use of advanced but feasible concepts in structure and equipment in order to meet necessary weight constraints. In particular, electronic and power supply equipment must be substantially lighter than equipment presently employed in Apollo.

4. For maximum packaging density and minimum vehicle weight in the lunar landing configuration, toroidal propellant tanks will be employed. This entails some manufacturing problems and propellant feed complexity, but these problems are considered surmountable.

5. Advanced super insulation techniques must be employed for the cryogenic propellants along with special attention to tank support design. A concept which is believed to be feasible is presented herein, and a test program is in progress which is intended to substantiate this feasibility.

6. The manned lunar landing mission and the lunar logistic mission should be considered simultaneously in order that minimum hardware duplication and program costs are incurred. Specifically, the landing stage of the cryogenic propelled manned landing spacecraft is adaptable with minor change

for use as the lunar logistic vehicle. This approach is recommended considering minimum total program cost, maximum hardware utilization, and program growth potential.

4.0 MISSION ANALYSIS

4.1 FLIGHT MECHANICS

For the purpose of discussion, the mission is treated in three phases: (1) Translunar Phase; (2) Descent Phase; (3) Ascent Phase. The characteristic velocity (ΔV) required for the various phases are shown in Table 4-I.

4.1.1 Translunar Phase

The translunar phase begins with translunar injection of the lunar mission system by the S-IVB. The nominal translunar trajectories result in transit times of 66 to 72 hours. Two mid-course corrections are made; the first one approximately 20 hours after translunar injection, and the second one 30 hours later. A final correction will be required during lunar orbit injection. The ΔV allowance of 300 fps for these three corrections is consistent with current Apollo ΔV requirement. However, Apollo studies indicate that this could be reduced to 150 fps while maintaining a better than 3σ probability level based on the following conditions:

1. C-5 translunar injection (MSFC platform).
2. 720-degree Earth orbit coast.
3. One hour sample rate.
4. Ten arc second sextant.
5. Two mid-course corrections.

4.1.2 Descent Phase

The descent phase begins when transfer to the landing orbit is initiated. This is accomplished by applying retro thrust approximately 180

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TABLE 4-I

MISSION ΔV REQUIREMENT

PHASE	ΔV - fps
TRANSLUNAR (66 - 72 hr transit time)	
Mid-course corrections	300
Lunar orbit injection (100 n.mi. circular orbit - land at any point within ± 10° of the lunar equator)	3,670
Reserve 5%	199
Translunar Phase Total	4,169
DESCENT	
Hohmann transfer to 50,000 ft perilune	125
Retro descent to h = 1000 ft, $V_v = 10$ fps and $\gamma = 90_o$ ($T/W_o = .465$)	5,820
Hover, translate, and land	800
Reserve 5%	337
Descent Phase Total	7,082
Translunar and Descent Phases Total	11,251
ASCENT	
Launch and inject into 100,000 ft circular orbit ($T/W_o = .55$)	5,935
Trans-Earth injection (60 - 64 hr transit time)	3,344
Mid-course correction	300
Reserve 5%	479
Ascent Phase Total	10,058
Mission Total	21,309

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degrees from the landing site. The resulting elliptical orbit will have a perilune of 50,000 feet. This maneuver requires a ΔV of 125 fps. The main retro descent to the landing site is initiated at perilune of the landing orbit. The ΔV allowance (5780 fps) for retro descent is based on the use of an optimum T/W_o and optimum steering during descent to the landing initiation point (h = 1000 feet, vertical velocity = 10 fps, horizontal velocity = 0). A ΔV of 800 fps has been allowed for hover, translation, and descent from the landing initiation point.

4.1.3 Ascent Phase

The ΔV allowance for ascent (5935 fps) is based on direct ascent to a 100,000 feet circular orbit along an optimum steering trajectory using the optimum T/W_o. This ΔV could be reduced by boosting to a low altitude and utilizing a Hohmann transfer to achieve the 100,000 circular orbit; however, the resulting low angle climb-outs would have to be analyzed with respect to flight control and terrain clearance. Using this method, the end boost altitudes and approximate required ΔV's are as follows:

h (end boost) ft.	ΔV (fps)
100,000	5935
50,000	5780
25,000	5750

Trans-Earth trajectories resulting in transit times of 60 to 64 hours will require a ΔV allowance of 3344 fps. The corresponding mid-course correction ΔV allowance is 300 fps. As in the case of the translunar mid-course corrections, this could probably be reduced without reducing mission reliability.

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Earth atmosphere re-entry for the above trajectory would be consistent with current Apollo re-entry constraints and capabilities.

4.1.4 Characteristic Velocity Reserve

At this time, there is some question about what constitutes a reasonable velocity reserve. Obviously, a number of factors affect the propellant required to accomplish a particular mission. Normal design procedure accounts for such factors as boil-off, engine mixture ratio shift, trapped propellants, etc. Therefore, the velocity reserve is intended to account for the increased ΔV requirements due to non-optimum flight profiles and deviations in propulsion system efficiency.

Since the Apollo crew will be capable of both monitoring and overriding the automatic flight control system, the maximum deviations from the optimum flight profile will be determined by the crew's ability to fly the required optimum profiles. With regards to propulsion system efficiency, an engine I_{sp} of 425 seconds has been used to determine propellant weight requirements. The Pratt & Whitney RL-10A-3 engine specification (Reference 1) calls out an I_{sp} of 424 seconds as a 3σ lower limit and 429 seconds as a nominal value. Hence, any deviations in engine performance should tend to reduce the required velocity reserve. Therefore, the reserve ΔV allowance of 5 percent (based on optimum flight profile requirements) is primarily intended to account for planned or emergency crew controlled maneuvers. This is considered to be a realistic reserve allowance for trained crews using a visual display-cue system type flight control scheme.

4.1.5 Propulsion System Throttle Ratios

Figure 4.1 shows the effect of T/W_o (based on Earth weight at phase initiation) on descent and ascent ΔV requirements. Also shown is the

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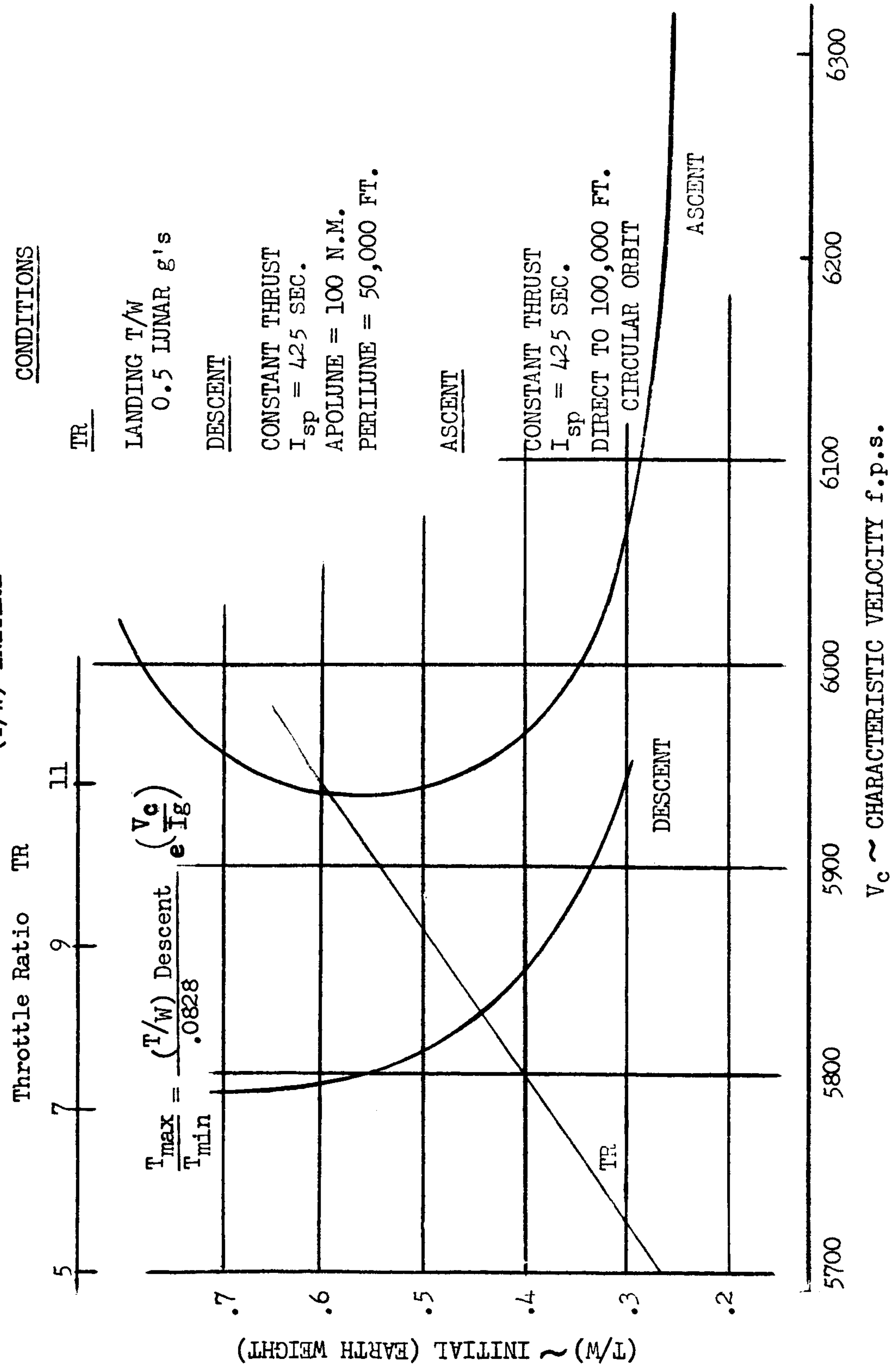
FIGURE 4.1

LUNAR DESCENT & ASCENT CHARACTERISTIC VELOCITY & THROTTLE RATIO REQUIREMENTS
VS
(T/W) INITIAL



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engine throttle ratio required to achieve a T/W = .5 lunar g's at initiation of the final landing maneuver. The effect of T/W_o at lunar orbit injection and trans-Earth injection has been evaluated and found to be negligible for T/W_o between .3 and .7.

4.1.6 Mission Abort and Emergency Landing Capabilities

Of primary importance are those areas in which the two-stage direct mission concept affects the abort and/or emergency procedures or capabilities of the spacecraft. Abort capabilities of the direct landing mission vehicle during translunar flight and lunar orbit injection are greater than those of the Apollo for the LOR mission due to the increased available ΔV.

Descent to the lunar surface is the most critical area in terms of abort capability for the lunar landing mission. The nominal descent to the lunar surface is accomplished as follows:

1. Main retro is applied at an altitude of 50,000 feet.

2. Maximum thrust is maintained during descent, along a preselected flight profile, to the landing initiation point.

3. At the landing initiation point (h = 1000 feet, vertical velocity = 10 fps, horizontal velocity = 0), the engines are throttled and the sink rate is driven to zero.

4. Translation and descent to the landing site are accomplished using the engine throttle to control sink rate and thrust vector orientation to control translation.

To evaluate the effect of flight control, guidance or other system failures, requires that specific hardware, monitoring systems, and over-ride capabilities be synthesized. Current Apollo efforts in the area of complete system simulation are indicative of the complexity of such an evaluation.

In general, engine failure represents the most severe test of the vehicle's ability to complete its mission or to successfully abort the mission. Therefore, a simplified analysis of the effects of the failure of one engine during the descent phase should indicate the most critical areas of failure and probable requirements for successful abort or landing following such a failure.

Descent along an optimum flight profile results in fairly shallow flight path angles, on the order of -15 degrees maximum, until the last few seconds of main retro. The corresponding pitch rate is on the order of .06 deg/sec. The landing engine gimbal limits (15 degree null point gimbal with ± 5 degree control range) allow the engine thrust vectors to be pointed through the c.g. when approximately 22% or less of the landing stage propellant remains in the tanks. This corresponds to approximately 63% completion of the main retro phase. If engine failure occurs prior to reaching this point in the trajectory, it will be necessary to shut down the good engine immediately and initiate the correct abort procedure.

Abort would be accomplished by first driving the pitch and/or yaw rates to zero using the RCS. Analysis indicates that the rotation induced by engine shut-down could be zeroed in approximately three seconds. The command module and landing stage would then be structurally uncoupled. If the RCS (4 x 100 lb) is used to physically separate the vehicles, prior to take-off engine ignition, it would require approximately five seconds to obtain clearance between the take-off engine and the landing stage. Another two seconds would be required for the take-off engine to reach maximum thrust, giving an abort total time of ten seconds. During this time, the maximum altitude loss would be approximately 1500 feet. If the take-off engine is

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used for separation, the separation time is reduced to approximately two to four seconds. However, the use of the take-off engine would increase the possibility of contact between the engine and landing stage during separation due to moments resulting from misalignment of the thrust vector. If the RCS system is used for separation after the transient pitch and yaw rates have been zeroed, the maximum pitch or yaw acceleration due to RCS thrust variations producing moments equal to 5% of the maximum control capability in a plane would be approximately .22 deg/sec^2. This would result in a command module rotation of 2.75 degrees during separation and would rotate the engine centerline at the engine face a maximum of 6.3 inches away from the centerline of the landing stage; the allowable maximum without contact is approximately twelve inches. This would seem to indicate that separation may be possible without the use of guide rails, etc., but a detailed analysis must be made to establish the exact requirements.

If engine failure occurs after the propellant has been depleted to the point at which the engine thrust vector can be pointed through the c.g., a decision must be made to continue the descent or to abort. If abort is selected, the engines are gimbaled to point the engine thrust vector through the c.g.; the RCS is used to orient the thrust vector; and climb-out is initiated with the propellant remaining in the landing stage. Physical separation of the command and landing stages is accomplished as previously outlined. If the decision is made to continue the descent, the engines are also gimbaled to point the engine through the c.g., and the vehicle has to be tilted over until the good engine is aligned with the trajectory. However, the propellant available for hover, translation, and landing is reduced due to the increased fuel consumption associated with the lower T/W descent.

Nominally, the engines will be gimbaled such that the thrust vectors are through the c.g. before the landing initiation point is reached. From this point on, the engines will be throttled to obtain the desired sink rate. If engine failure occurs during the final landing maneuvers, either a single engine landing, or a single engine climb-out and abort can be made. However, there is a critical lower altitude which, when coupled with minimum landing stage propellant remaining, makes a single engine landing mandatory. This would occur when the engine failure altitude, plus the altitude gain possible with the remaining landing stage propellant, is less than the minimum altitude required for safe separation and abort. The actual minimum conditions for abort will have to be determined by a detailed analysis.

A single engine landing can be made in the following manner. At an altitude of approximately 50 to 100 feet, the translational velocity is driven to zero. Ath this time, the vehicle lognitudinal axis is canted from the vertical at an angle equal to the engine gimbal angle. The vehicle longitudinal axis is rotated to the vertical, a steady state roll about the longitudinal axis is established with the RCS, and the desired rate of descent is achieved by throttling the engine. The roll is required to limit the translational velocity at touchdown and to minimize the translation range during this final maneuver. The maximum translational velocity during this type of landing is given by the expression:

$$V_{max} = \frac{2\ g\ \tan\delta}{\eta\ \omega}$$

where

$g = 5.314$

δ = engine gimbal angle

ω = roll rate about longitudinal axis

$\eta = W/T \cos \delta$

W = lunar weight

T = engine thrust

The maximum resultant translational plus rotational velocity experienced by a landing leg is:

$$V_R = \frac{2g}{\eta} \frac{\tan \delta}{\omega} + \ell \omega$$

where

ℓ = length of landing leg measured from, and normal to, the vehicle centerline.

Figure 4.2 shows the relationship between V_R and ω for a constant sink rate ($\eta = 1$) and $\delta = 15$ degrees.

The performance of a single engine landing under manual or automatic control will have to be evaluated with regard to practical usage and reliability.

4.2 AERODYNAMIC HEATING

During its ascent through the Earth's atmosphere, the lunar mission systems studies vehicle will require some form of external thermal protection in order to control the temperature rise of the aluminum sub-structure within the acceptable limits. It was anticipated that the lunar landing stage, which mounts further aft and has a cylindrical configuration, will have lower heat fluxes and, therefore, would not require thermal insulation. A preliminary aerodynamic heating analysis was conducted to (1) assess the insulation requirements on the forward vehicle and, (2) predict the transient temperature profile on an un-insulated wall of the aft vehicle.

The highest heating rates during boost will occur at the most forward conical wall position just aft of the Apollo command module. This point

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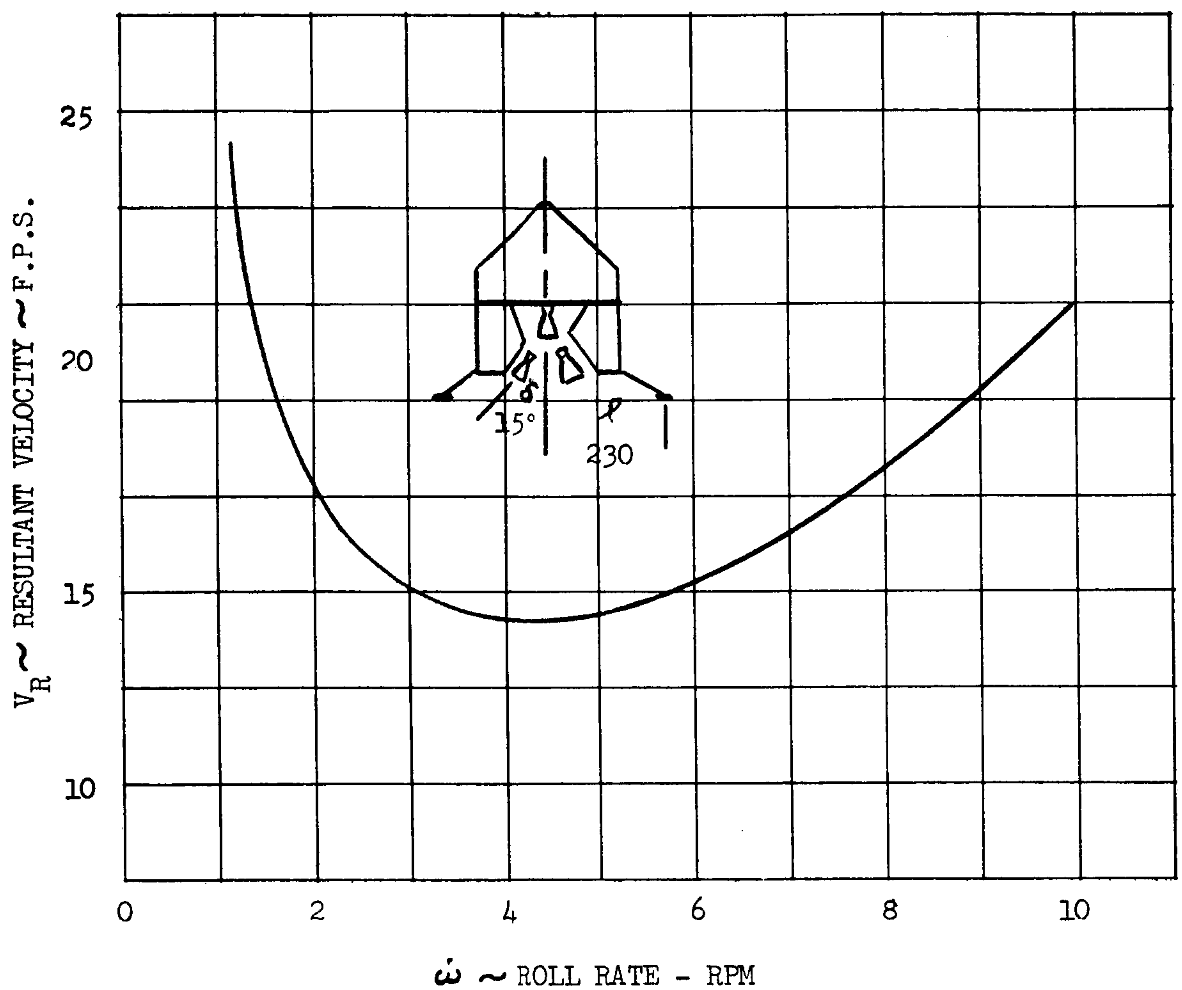

FIGURE 4.2

MAXIMUM RESULTANT VELOCITY VS ROLL RATE

was, therefore, selected for the heating analysis of the forward vehicle.

Boost heating rates and wall temperature profiles were determined for the insulated and un-insulated forward fairing, Figures 4.3 through 4.5 of the lunar mission system vehicle. The un-insulated wall temperature profile of the cylindrical section of the lunar landing stage is shown in Figure 4.6.

Due to the relatively high boost heating rates and a low allowable working temperature, the aluminum forward fairing would require the use of a low temperature ablation material. This design would provide adequate thermal protection for the polyurethane foam filler material used for meteorite protection which is limited to approximately 1000°R. Results of the analysis indicated that a 0.10 inch thick layer of cork which ablates at 360°F would adequately insulate the aluminum sub-structure and meet the ablation rate requirements. The amount ablated would be 0.032 inches or 32% of the initial thickness. In view of this, the thickness could be reduced, but this would result in higher sub-structure temperatures. The maximum predicted aluminum wall temperature just aft of the Apollo command module would be 680°R (220°F) for the insulated wall and 1258°R (798°F) for the un-insulated case.

However, for reasons which will be outlined in detail in Section 9, a double-wall sheet stringer skin was selected with Rene 41 as preferred sheet material and no insulation.

4.3 GUIDANCE AND NAVIGATION

One of the most critical factors in the design of a manned space-

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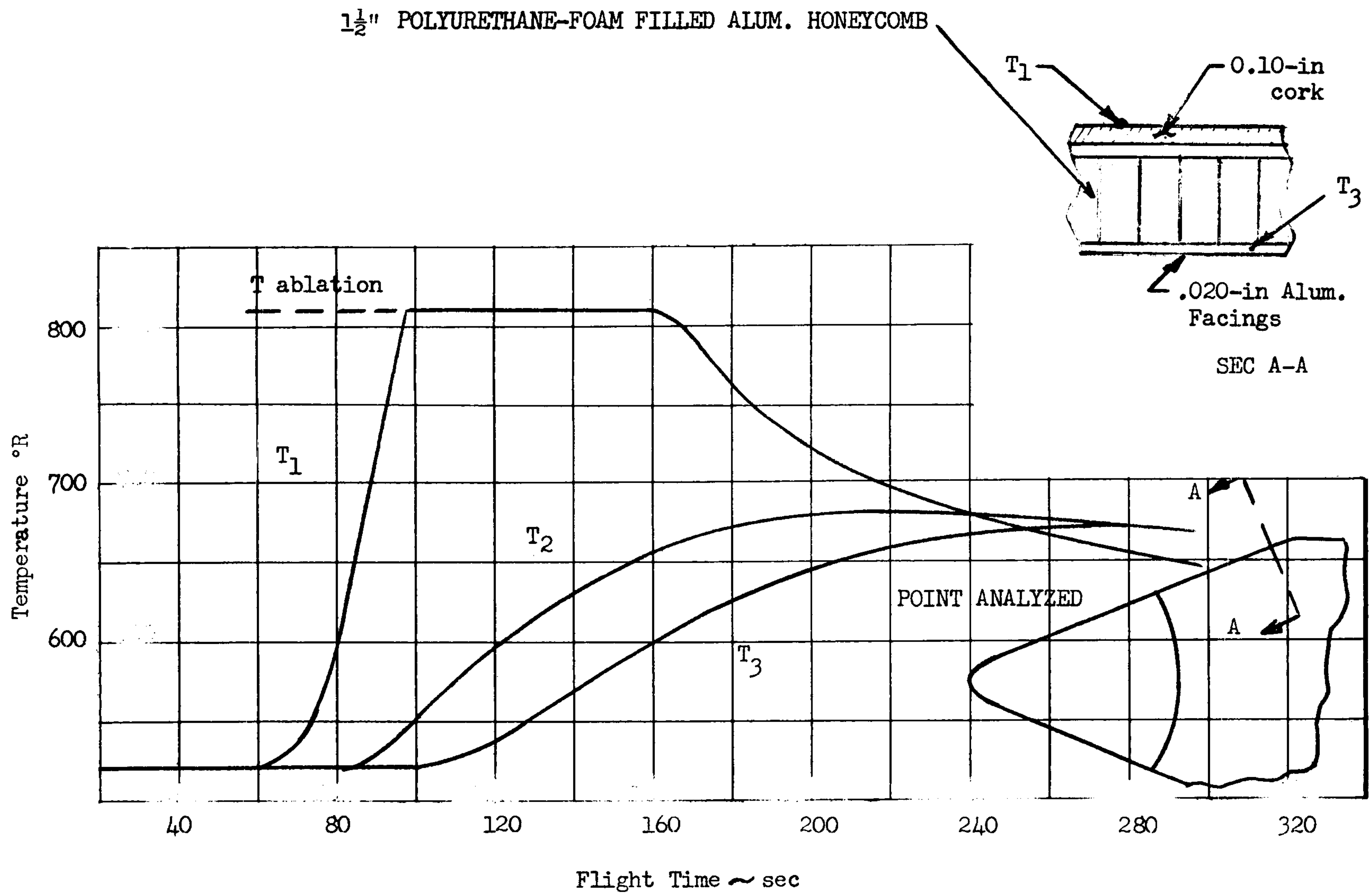

FIGURE 4.3

TEMPERATURE HISTORY AT FORWARD WALL LOCATION DURING BOOST

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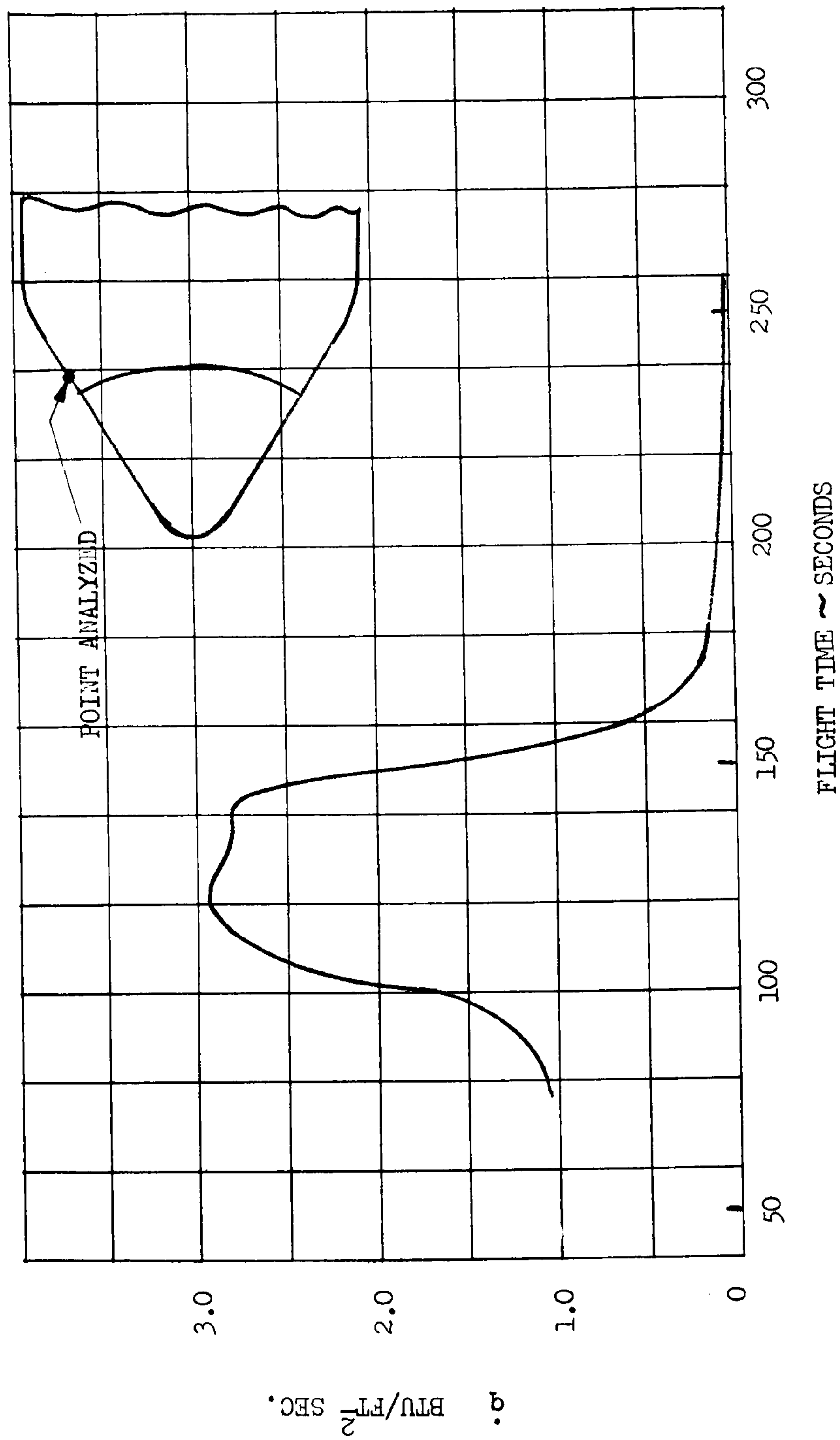

FIGURE 4.4

HEAT FLUX HISTORY AT FORWARD FAIRING DURING BOOST

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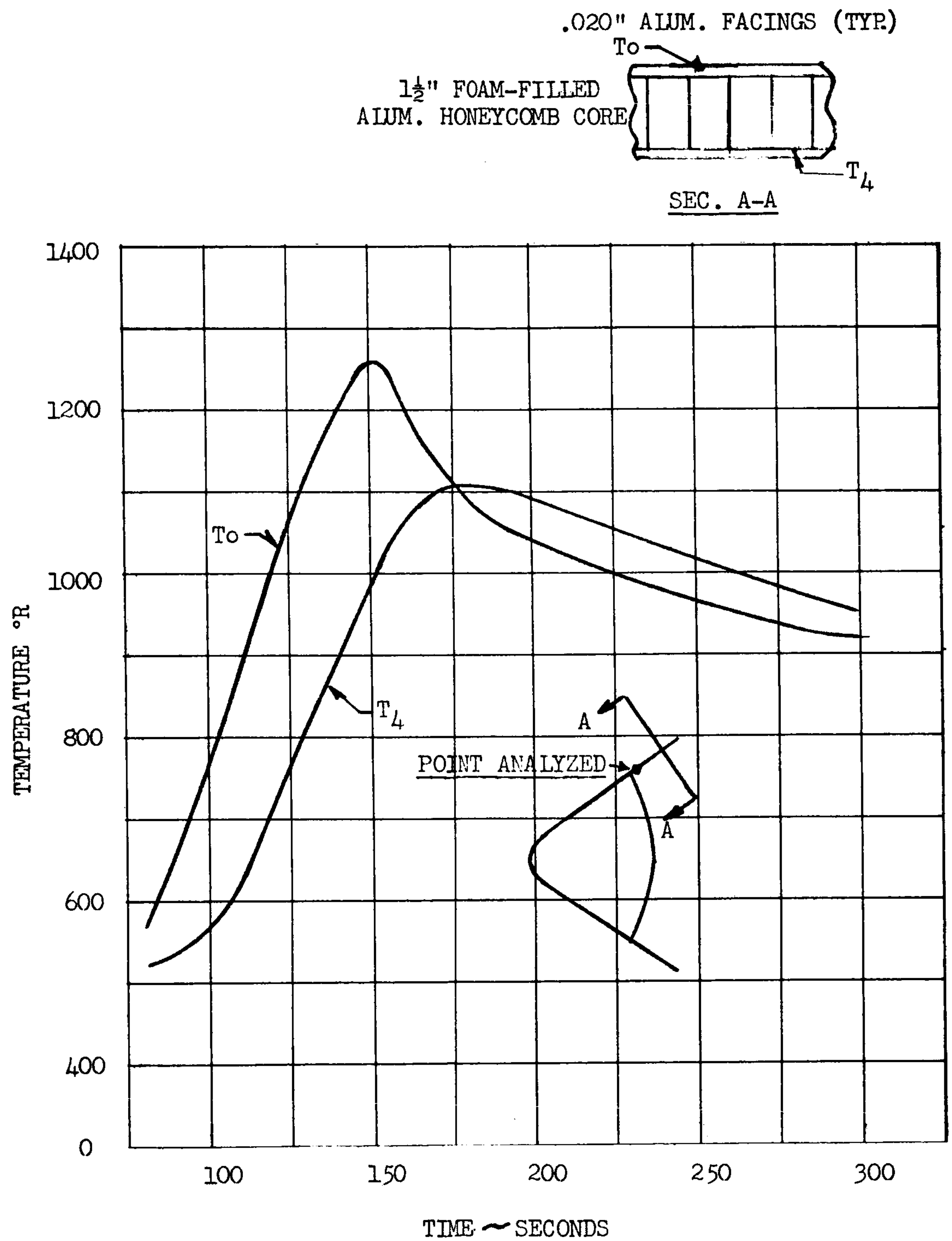

FIGURE 4.5

TEMPERATURE HISTORY OF UN-INSULATED FORWARD WALL DURING BOOST

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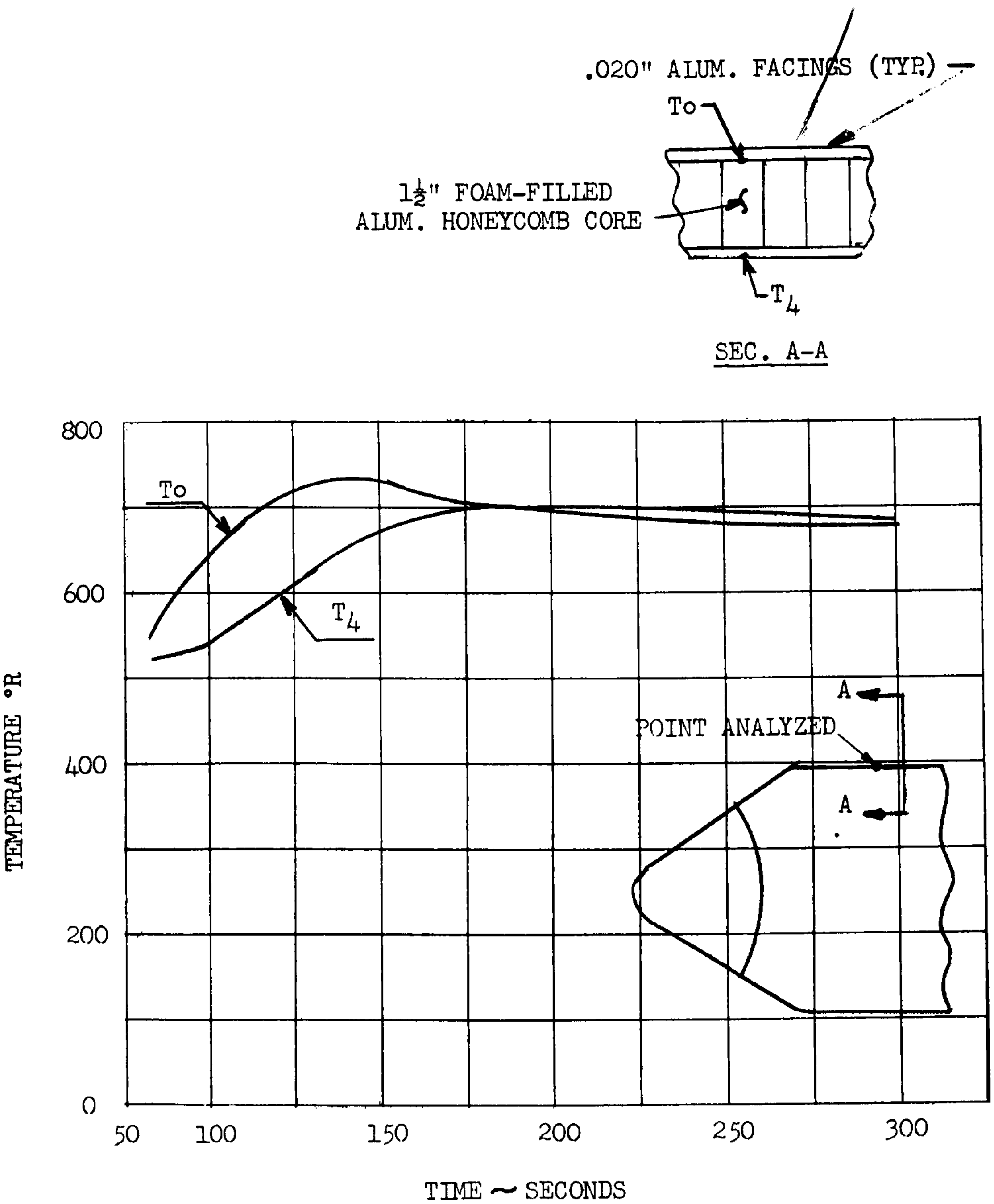

FIGURE 4.6

TEMPERATURE HISTORY OF UN-INSULATED CYLINDRICAL WALL DURING BOOST craft is the reliability of the spacecraft and its equipment. Since the guidance system is one of the most complex of the spacecraft subsystems, special attention to reliability is required. Generally, reliability is enhanced by use of simple equipment concepts. In a manned operation, the system can be designed to make optimum use of human judgment, thereby increasing the reliability. It is for this reason that the philosophy followed in meeting the study objectives is to increase human participation, wherever possible, during all phases of the lunar mission.

Of particular importance is the utilization of man's decision-making ability and versatility in performing a variety of navigation data-taking functions onboard. This method is possible because throughout most of the flight there is ample time for taking data, and for checking and cross-checking the results. This concept can be translated into hardware by replacing fully atuomatic functions by manual or semi-automatic. This reduces the complexity of the equipment, increases reliability and flexibility, and reduces the weight and power requirement.

4.3.2 The Guidance System

In order to obtain the latest information in the field of components and systems, a variety of manufacturers were contacted and informal meetings were held with their technical representatives. This, together with a review of a variety of other literature, allowed a survey of the technology in both present day, and projected future, guidance and navigation hardware.

The Guidance and Navigation system shown in Table 4-II has been proposed as a result of this study.

This table is a tabulation of the proposed guidance and navigation

TABLE 4-II

PROPOSED GUIDANCE AND NAVIGATION SYSTEM

COMPONENT	Weight (lb)		Power (watts)	
	Miniature	Subminiature	Miniature	Subminiature
IMU (4-axis gimbal assembly)	28	16	129	100
Platform electronics	24	18	196	150
Digital coupler	30	20	178	130
Computer and power supply	54	28	150	40
Sextant	5	5		
Base (auxiliary platform)	18	12	40	40
Star catalogs and maps	5	5		
Theodolite	10	10		
Radar altimeter	35	25	200	150
*IRU (3 gyros and 3 accelerometers)	17		45	
*Computer (coordinate transformation)		5		7
**TV camera and electronics		6		6.5
**TV monitor		8		15
**Telescope (periscope)		25		
Sub-total	209	139	993	610
Incidentals	11	11		
TOTAL	220	150		

*These items represent equipment used by Navigation and Guidance as a back-up inertial system. However, these items also are used by Stabilization and Control; hence, although listed here for completeness, these items are accounted for in the SCS list.

**These items represent equipment mainly used by the communications and instrumentation systems; therefore, they are accounted for in the C. and I. list.

hardware with indications of approximate weights and power requirements. It is of importance to note that, while both miniature and subminiature components are listed, it is more realistic to assume that the former will be employed. The latter have been listed for purposes of comparison and as an indication of the various values that are being quoted for use in the near future. While the subminiature components are being quoted by every manufacturer as being the state-of-the-art in the near future, indications are that these components will be available somewhat late for use in the direct lunar mission.

4.3.3 Mission Phases

Following is a brief description of the proposed Guidance and Navigation system tasks during the various phases of the direct lunar mission.

During launch from either the Earth or the Moon, precise inertial guidance is required. While radio guidance could be considered for the Earth launch, there is little prospect for the use of radio guidance during the lunar launch. Nevertheless, radio guidance (or ground-based guidance) always will be considered as a back-up system, due to its availability. During the application of the velocity corrections (mid-course), there is a requirement for inertial measurements to monitor both the magnitude and direction of the velocity correction.

The IMU consists of a gyro-stabilized inertial platform which serves as a mounting base for three linear accelerometers. It is necessary that the platform be aligned relative to a known coordinate system. The stars furnish the most convenient source of reference. Since this is impractical during launch, accurately aligned gyros supply the required inertial memory. The IMU is aligned in space by sighting on a few stars displaced by 45 to 135 degrees. The sighting of the stars will be done with the manual sextant. The sextant is mounted on an auxiliary platform which may be either slaved to the main platform through some synchros, or consist of a small secondary platform. The sextant will be used for subtended angle measurements and for alignment of the inertial platform.

The use of the slaved platform greatly simplifies the alignment problem, because the sectant angles, relative to the platform, can be preset. The sextant also will be used during mid-course by observing occultations, and by measuring the subtended angle between a predetermined star and the limb of either Earth or Moon. The auxiliary platform is useful because it physically facilitates the performance of the actual manual measurement, and it also allows for compensation of any refraction error which may occur as a result of light traveling between life-supporting atmosphere inside the capsule and the vacuum outside. This platform facilitates the measurement of the angle of incidence to the window, so that compensating corrections can be performed.

It is also helpful to have onboard a manually-operated theodolite mounted on the same auxiliary platform. This will allow measurements of the direction to the edge, center, or prominences on the surface of the Earth or Moon. The reticle of this theodolite consists of a series of concentric circles, thus allowing accurate sighting on the center of the Earth or Moon. Stadiometric ranging also is possible with the help of this simple device.

To make use of the navigation measurements during mid-course, trajectory determination calculations will be performed. Each measurement, or set of measurements, need not determine the position or velocity of the spacecraft. Rather, a series of partial fixes taken over a period of time can be used to determine the trajectory of the spacecraft. Statistical methods using various recursive forms for performing a least square fit to the available trajectory data may be necessary to reduce the significance of random errors in the measurements. Hence the results of the navigation measurements will be applied to the vehicle-borne computer and the computer will determine the required ΔV corrections as a function of time. The digital computer considered for this mission is a general purpose serial machine with built-in redundancy and a storage capacity of about 8000 words, with approximately 24 bits per word.

It is intended to employ the man's decision-making ability in determining when to abort the mission and when to perform a ΔV correction. This may be done in the following manner. Generally, the amount of fuel required to perform a mid-course correction is increasing monotonically with time, while the available fuel onboard to perform a mid-course correction is fixed. Also, the uncertainty in the required ΔV is a decreasing monotonic function of time; therefore, a decision or trade-off exists. Remembering that there exists a one-to-one transformation between available fuel and the amount of velocity this fuel can produce, it is possible to govern the ΔV application by presenting the pilot with a display of the ratio of ΔV required to ΔV available as a function of time. Of course, as this ratio approaches unity, the abort condition is being approached.

The following closed-loop mid-course guidance philosophy may be followed: Upon determining in advance the time (t_C) when a mid-course correction is to be performed, a suitable target vector for guiding the vehicle is the required velocity ($\overline{V}_R$). This vector is determined by the mid-course correction sub-routine in the orbit determination computer and is a fixed quantity at the time of correction. Also, from the orbit determination computer, the vehicle's velocity ($\overline{V}_m$) is obtained at the same time. The difference between $\overline{V}_r$ and $\overline{V}_m$ is then computed, obtaining the velocity to be gained ($\overline{V}_g$). Since both m_{t_C} and $\overline{V}_g$ are known some time before the actual correction is made, the operator has time to align the vehicle so that the mid-course correction engines, when fired, will produce a thrust acceleration in the approximate direction of $\overline{V}_g$. Steering rates may then be based on the small angle between a T and $\overline{V}_g$, which will result in small steering rates. During thrust application for velocity correction, the thrust acceleration can be measured by an inertial system. The output of the three accelerometers can be integrated, thus monitoring $\overline{V}_g$ and $\overline{V}_m$. When the angle between $\overline{V}_R$ and $\overline{V}_m$ is reduced to zero, and $\overline{V}_R = \overline{V}_m$ (i.e., $\overline{V}_g = 0$), the engines are cut and the mid-course correction is ended. This process may be repeated as often as necessary.

This method assumes that $\overline{V}_R$ does not change appreciably in the vicinity of time t_C. This implies that $\overline{V}_g$ is applied impulsively, not allowing the vehicle to change position during that time. For all practical purposes this is true, because the mid-course corrections are small and, therefore, of small duration, which results in a small change in position and $\overline{V}_R$. In order to compute $\overline{V}_R$ continuously, a four-body, high-speed computer is required; this is not justifiable in this case.

The orbit determination is performed on a discrete time basis, so that continuous calculations of $\overline{V}_R$ are not required. The mid-course corrections could also be performed on a completely open-loop basis without losing too much accuracy. However, since an onboard computer is available from the launch guidance, a small modification of this computer justifies the choice of the closed-loop mid-course guidance described above.

Prior to lunar landing and during hovering maneuvers, both the radar altimeter and the TV system will be used. The TV system is available onboard for various functions, as indicated under Communication and Instrumentation. The availability of this system makes it possible to use the camera attached to the telescope (periscope) for purposes of surveying the lunar surface and in choosing the exact landing area. Thus, this operation will end at actual lunar touchdown. The TV monitor will be displayed so that all crew members may make use of it during manual control of the vehicle.

In view of the limitations imposed by radio blackout and vehicle maneuvering during re-entry, inertial guidance seems to provide the only practical means for achieving this objective. Since the initial conditions provided by the mid-course guidance system at the start of re-entry are sufficiently precise, the re-entry can be achieved with the inertial instruments listed in Table 4-II.

The inertial reference unit system provides the capability for back-up guidance, if the main inertial guidance system fails. This unit consists of three rate-integrating gyros and three floated accelerometers, and it functions as a simple strapped-down system. This system also uses a small computer for purposes of integration and coordinates transformation between inertial and spacecraft coordinates. Since this system is used by Stabilization and Control, a more complete description is given in this section.

A generalized block diagram of the proposed Guidance and Navigation system is shown in Figure 4.7.

Reference 1: RL-10A-3 Model Specification No. 2272D, Pratt & Whitney Aircraft. (15 August 1962) Revised 4 October 1962

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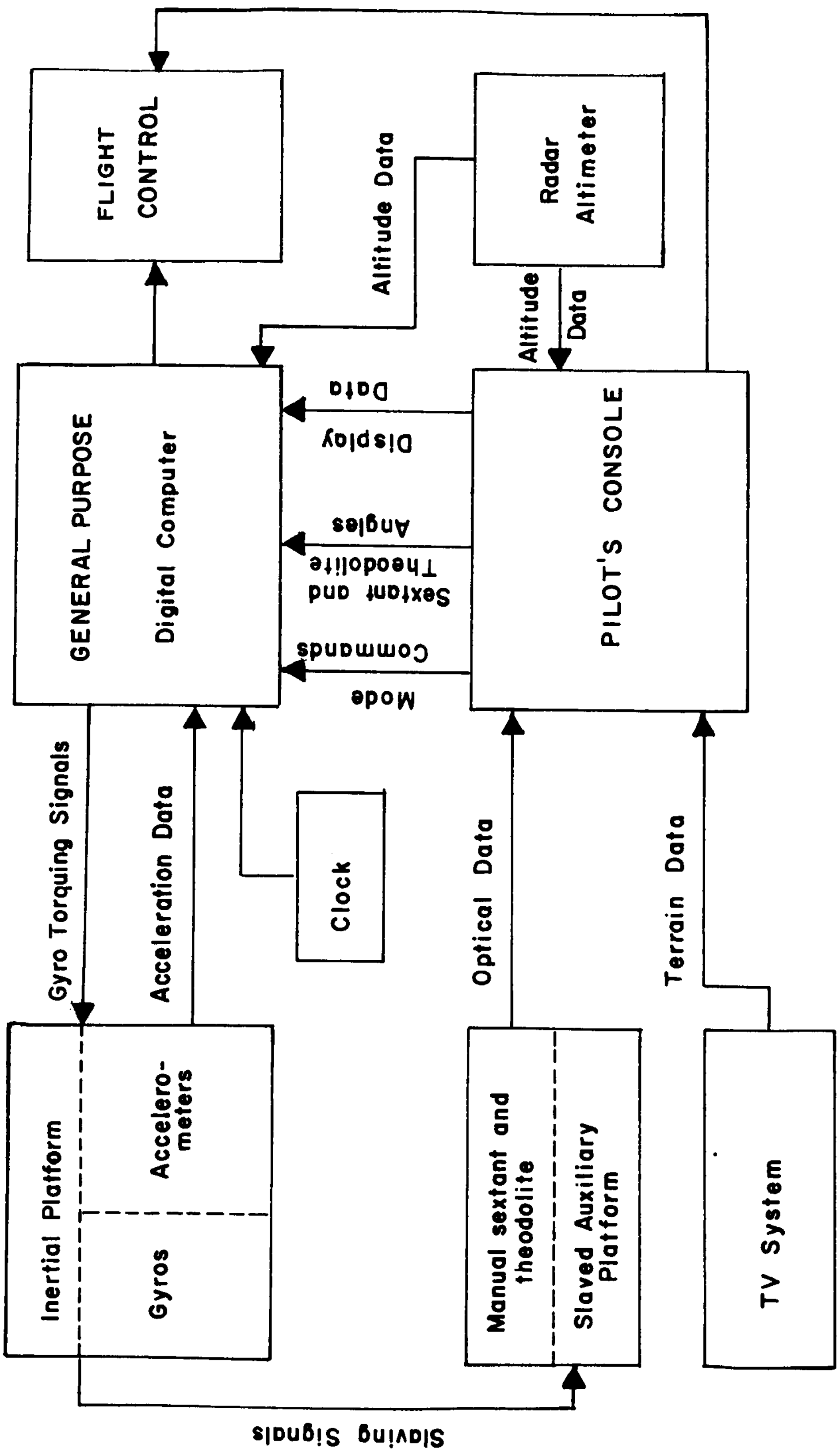

Figure 4.7 Generalized Block Diagram of Guidance and Navigation System

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5.0 PRELIMINARY DESIGN

The approach taken in this study was to optimize a three-man, fourteen-day direct lunar landing vehicle, using the Apollo command module with cryogenic lunar landing and lunar launch stages, compatible with the Saturn C-5 vehicle capabilites.

The selection of a basepoint design vehicle arrangement was the result of an evolution process of configuration development. The configurations investigated in the initial comparison are described in Reference (1). Of the configurations investigated, a basepoint design was chosen for further development and has evolved to the configuration shown in Figure 5.1.

This configuration, a two-stage vehicle, consists of a lunar landing stage, a lunar launch stage, and the Apollo command module.

5.1 GENERAL ARRANGEMENT

The structural arrangement of the selected vehicle configuration is shown in Figure 5.2. This vehicle consists of a 260-inch diameter cylindrical aluminum shell structure containing the lunar landing stage and a conical steel structure containing the lunar launch stage.

5.1.1 Lunar Landing Stage

The lunar landing stage for the direct lunar landing mission will utilize the Lunar Logistic Vehicle, Reference (2), with minor modifications. The payload adaptor which houses the reaction control system, the electrical power system, and the electronics equipment will be removed and the lunar launch stage will be attached to the primary structure of the landing stage. The electronic equipment that is removed from the Lunar Logistic Vehicle is not required for the direct mission as all of the functions will be accomplished

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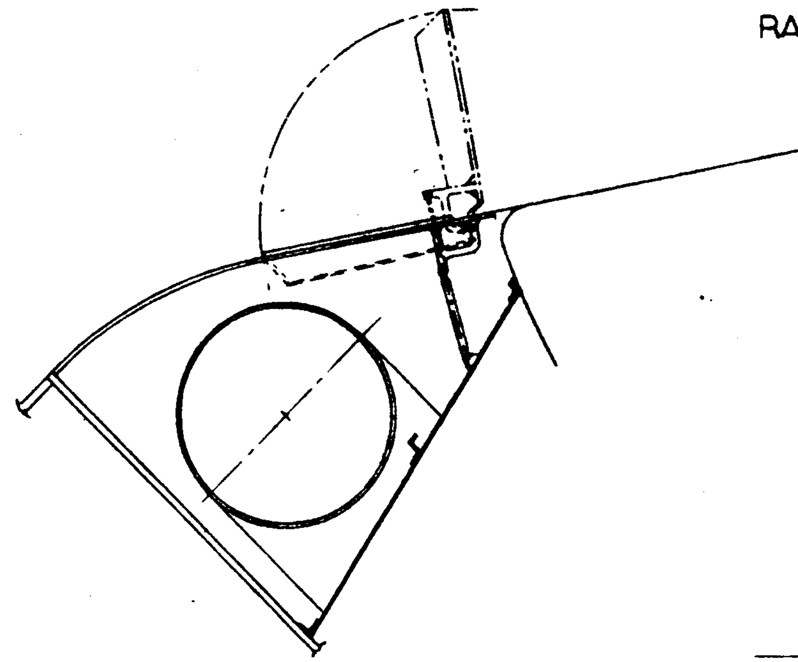
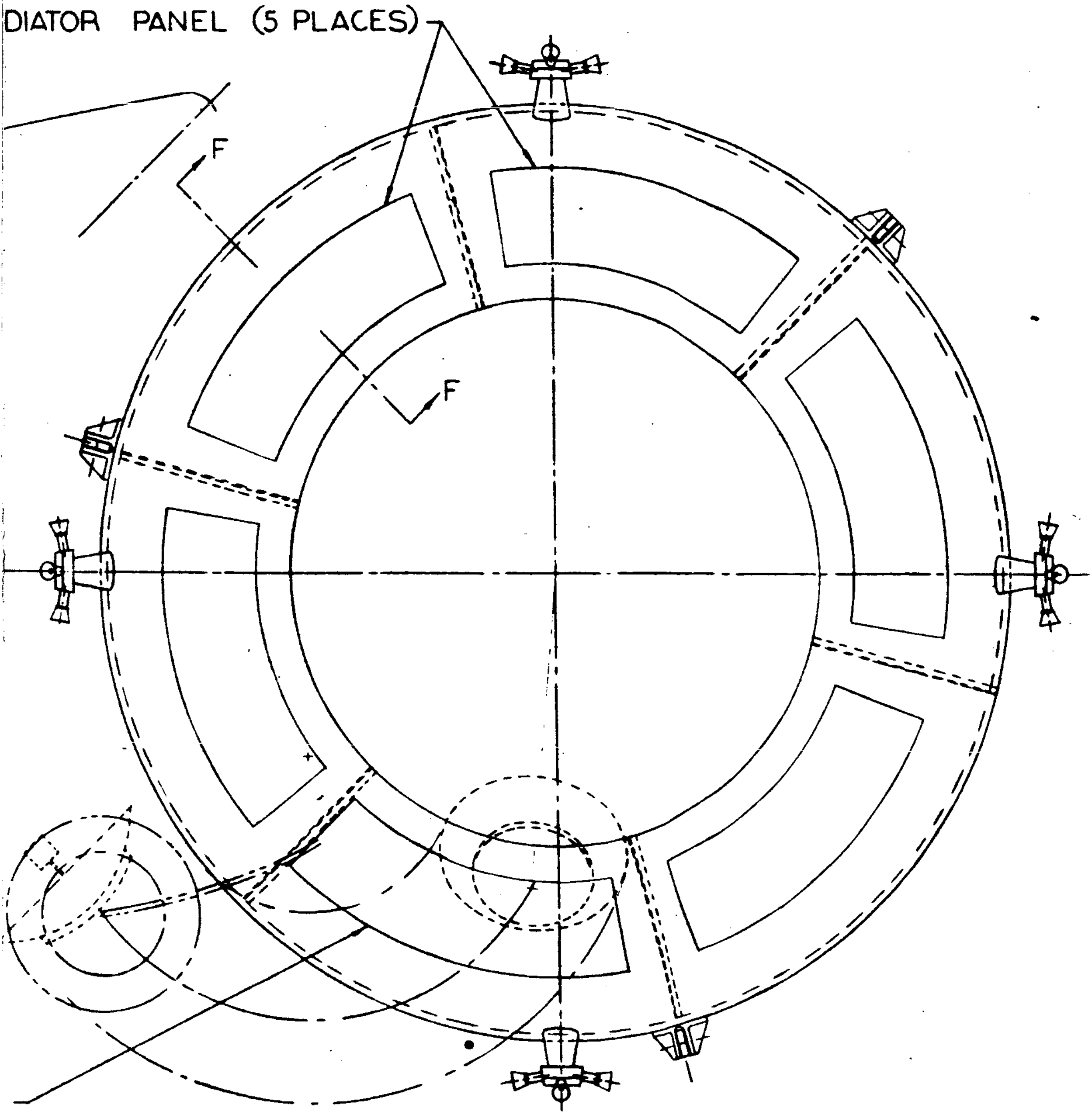


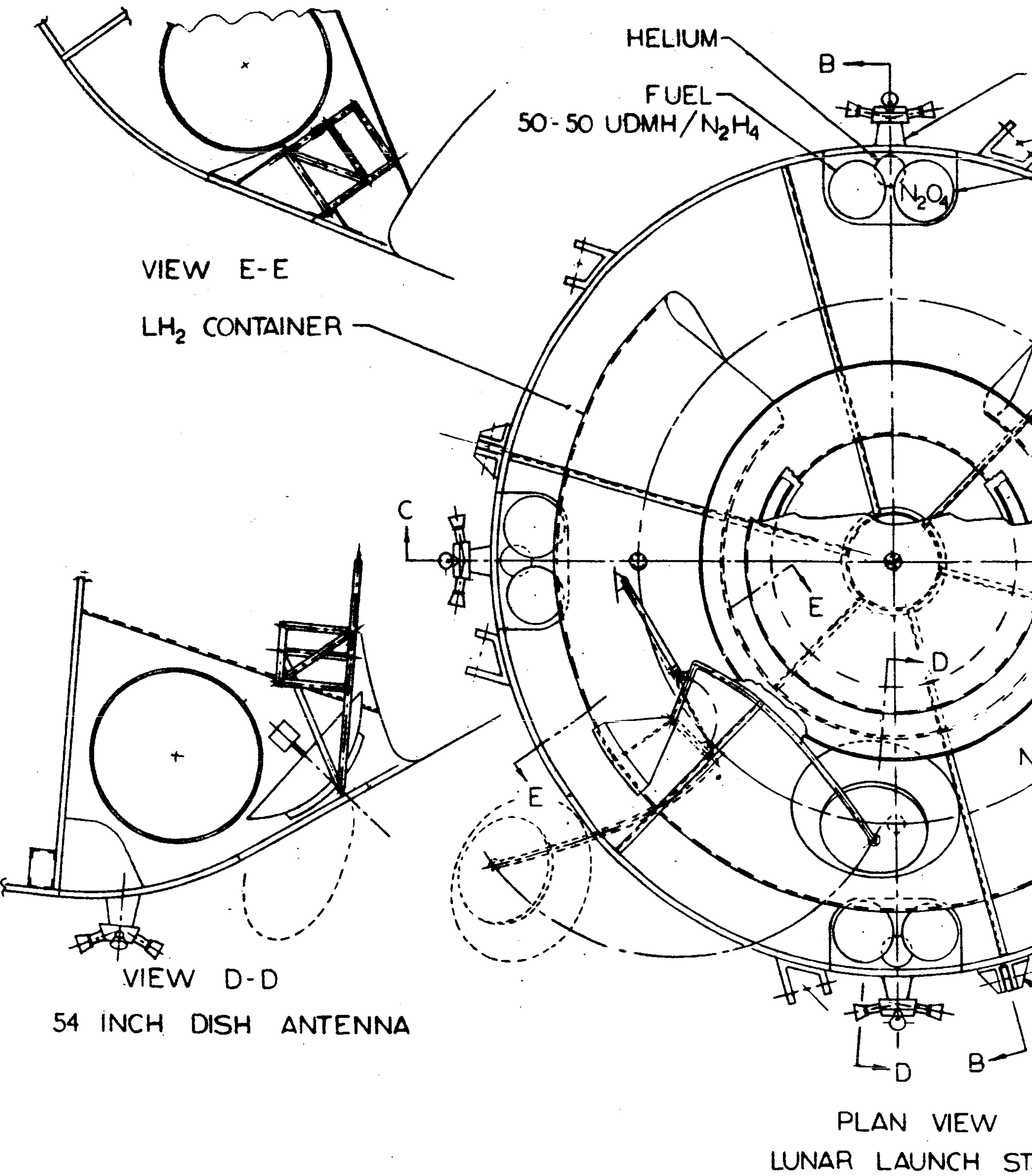
VIEW F-F
RADIATOR INSTALLATION

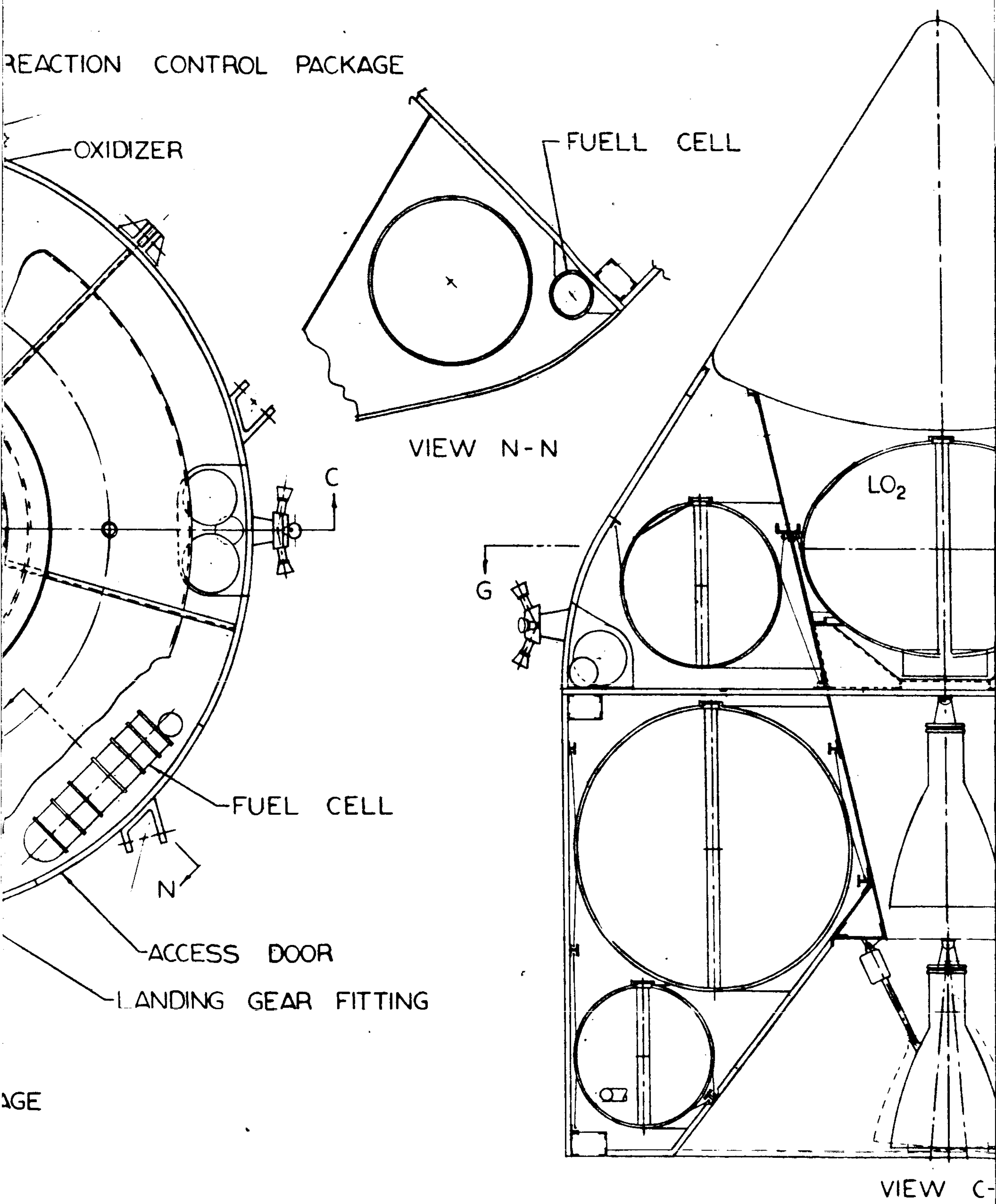
RA

ANTÉNNA DOOR

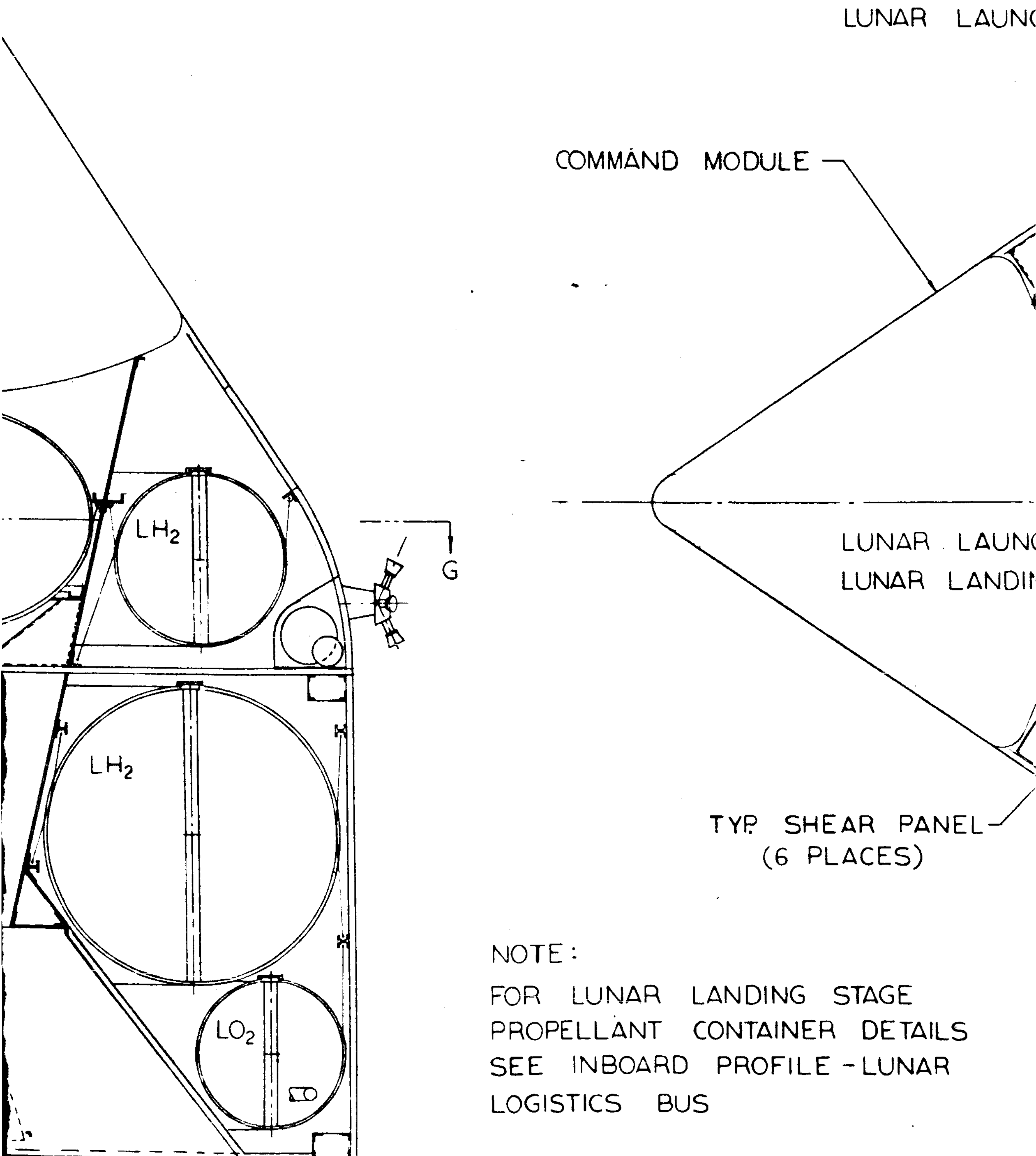


PLAN VIEW

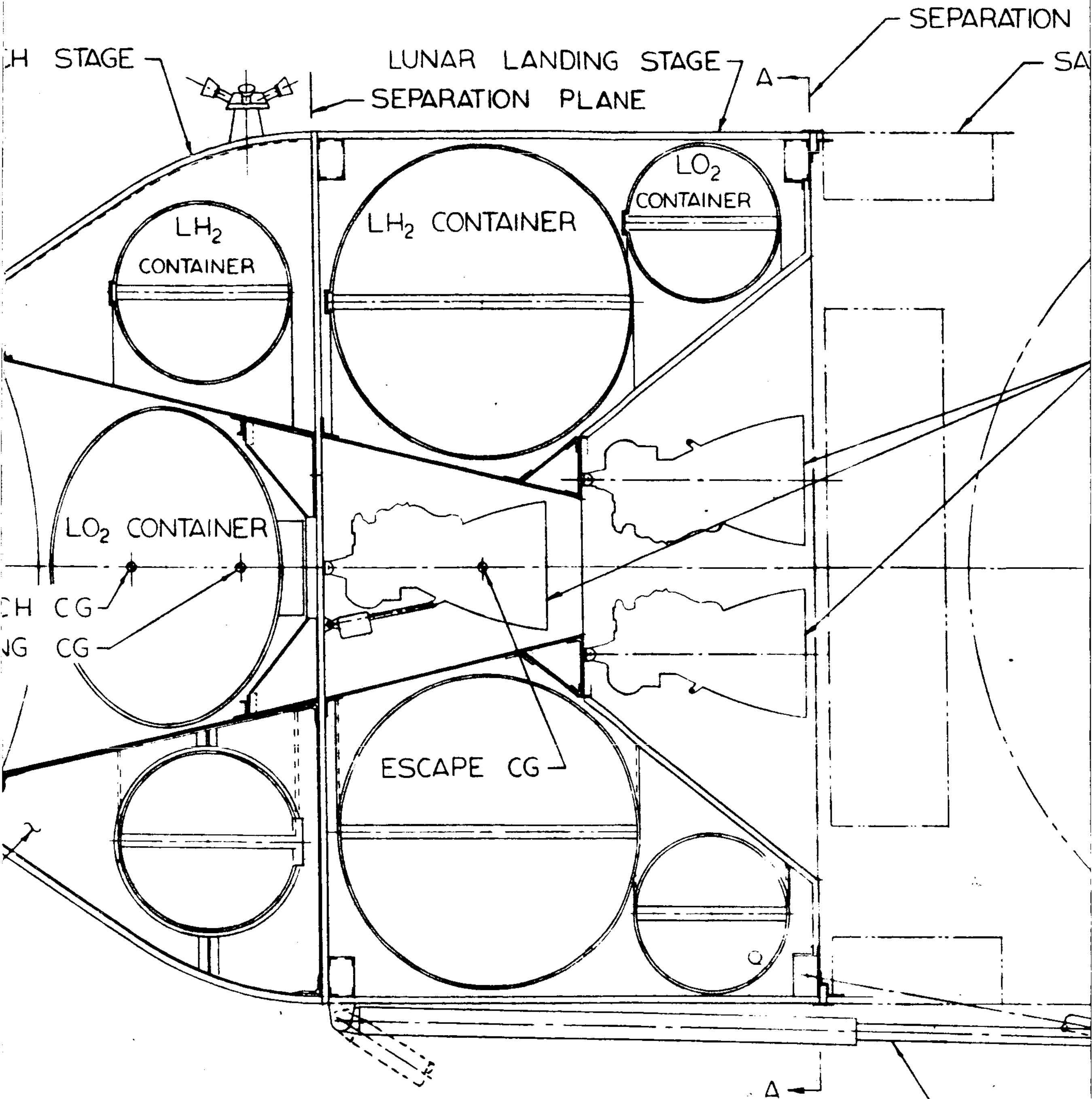

HELIUM
B
FUEL
50-50 UDMH/N_2H_4
N_2O_4
VIEW E-E
LH_2 CONTAINER
C
E
D
VIEW D-D
54 INCH DISH ANTENNA
B
D
PLAN VIEW
LUNAR LAUNCH ST


REACTION CONTROL PACKAGE
OXIDIZER
FUELL CELL
VIEW N-N
C
LO_2
G
FUEL CELL
N
ACCESS DOOR
LANDING GEAR FITTING
AGE
VIEW C-

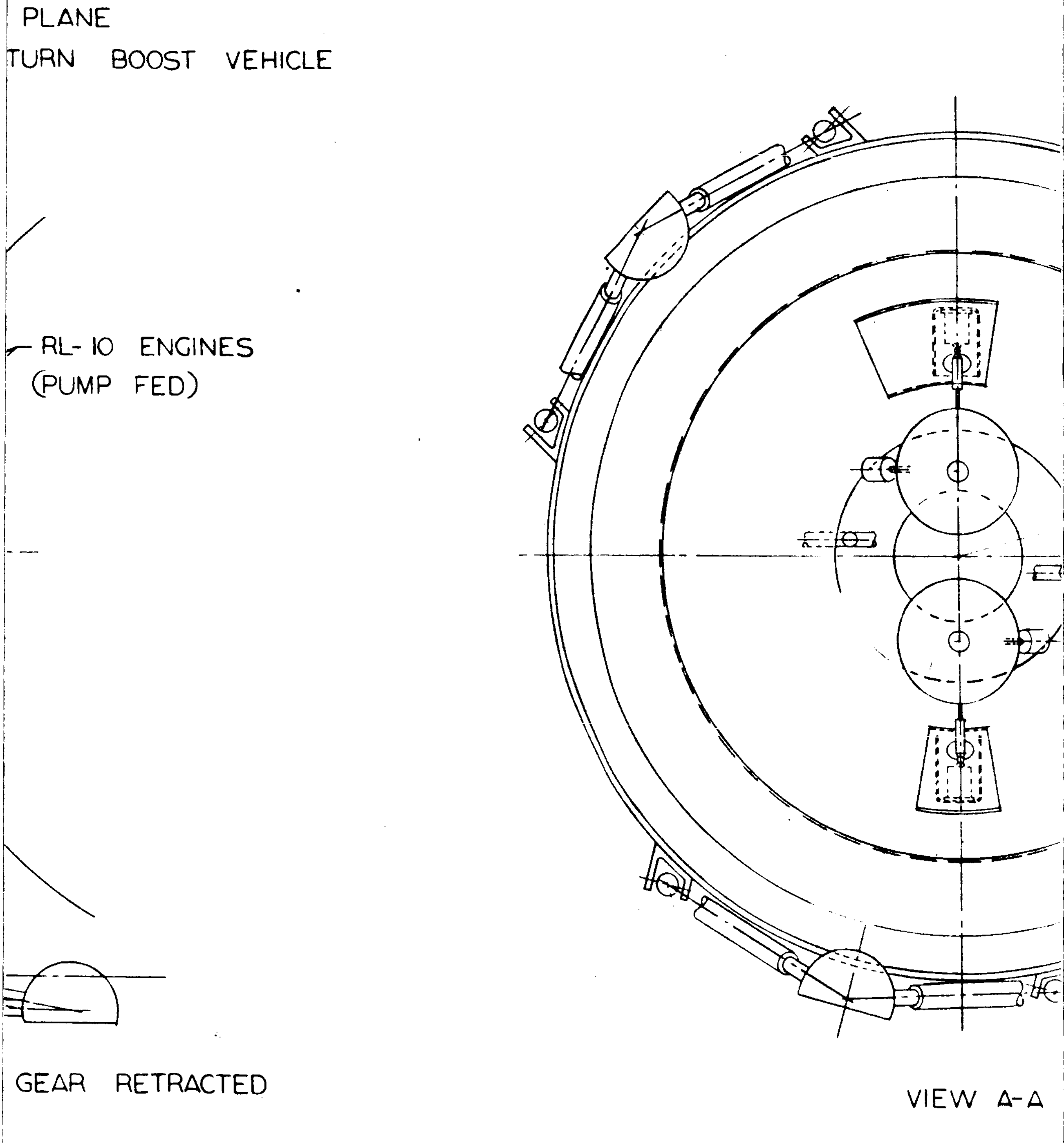
3

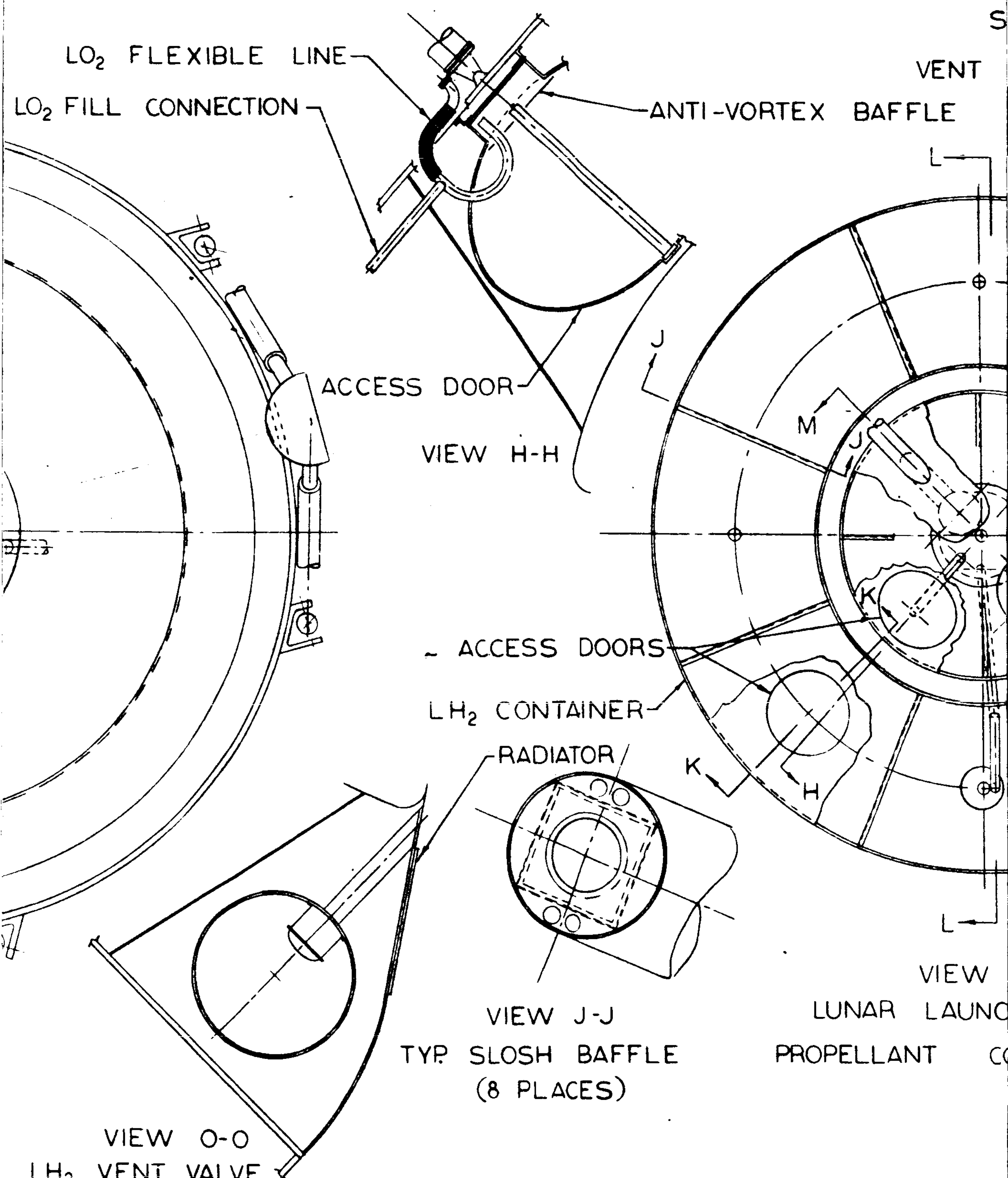


NOTE:
FOR LUNAR LANDING STAGE PROPELLANT CONTAINER DETAILS SEE INBOARD PROFILE - LUNAR LOGISTICS BUS


SEPARATION
H STAGE
LUNAR LANDING STAGE
A
SA
SEPARATION PLANE
LO_2 CONTAINER
LH_2 CONTAINER
LH_2 CONTAINER
LO_2 CONTAINER
H CG
NG CG
ESCAPE CG
A
LANDING

VIEW B-B


PLANE
TURN BOOST VEHICLE
RL-10 ENGINES
(PUMP FED)
GEAR RETRACTED
VIEW A-A


S
VENT
LO2 FLEXIBLE LINE
LO2 FILL CONNECTION
ANTI-VORTEX BAFFLE
L
J
ACCESS DOOR
VIEW H-H
M
J
K
ACCESS DOORS
LH2 CONTAINER
RADIATOR
K
H
L
VIEW
LUNAR LAUNC
PROPELLANT C
VIEW J-J
TYP. SLOSH BAFFLE
(8 PLACES)
VIEW O-O
LH2 VENT VALVE

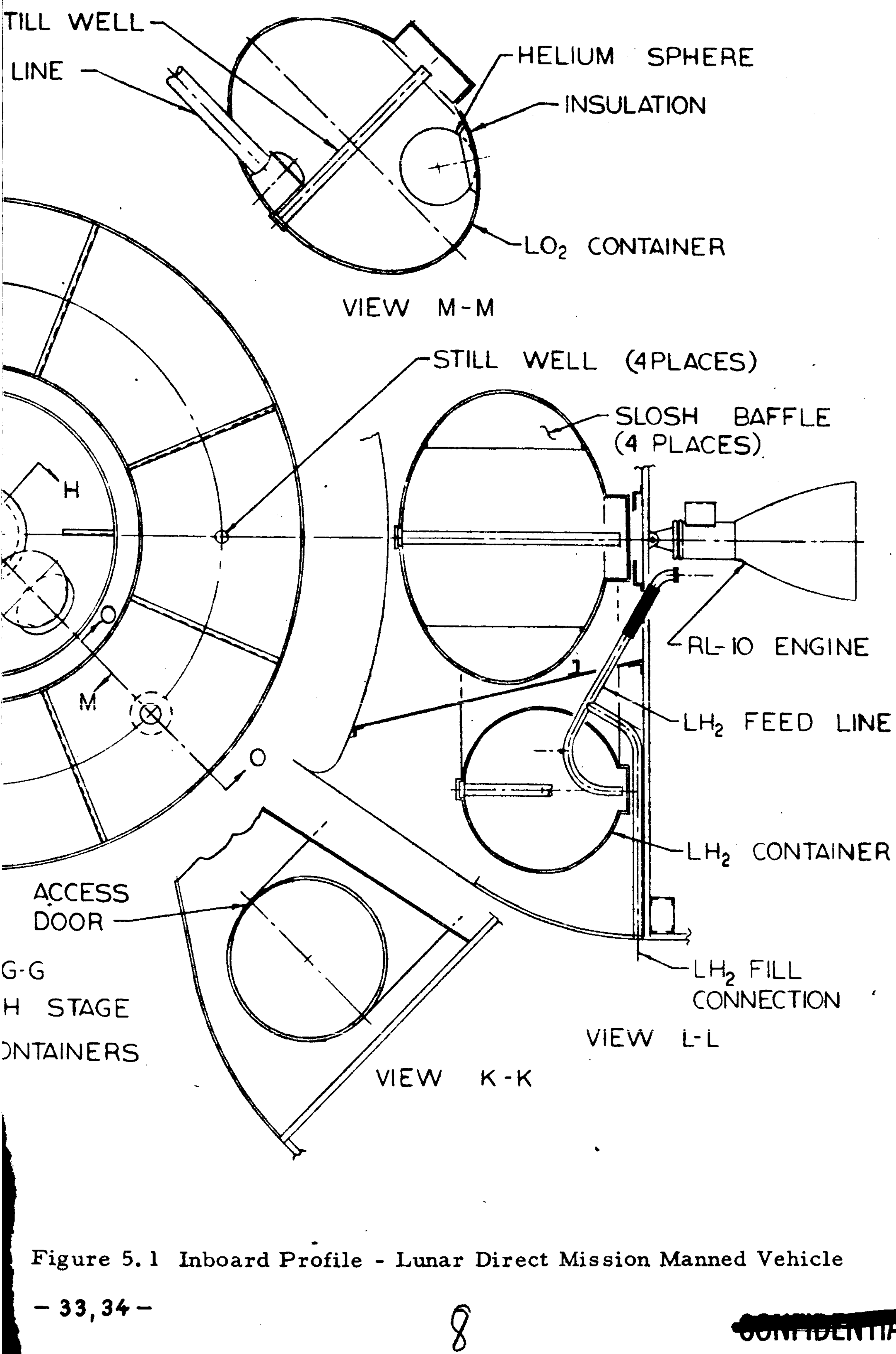
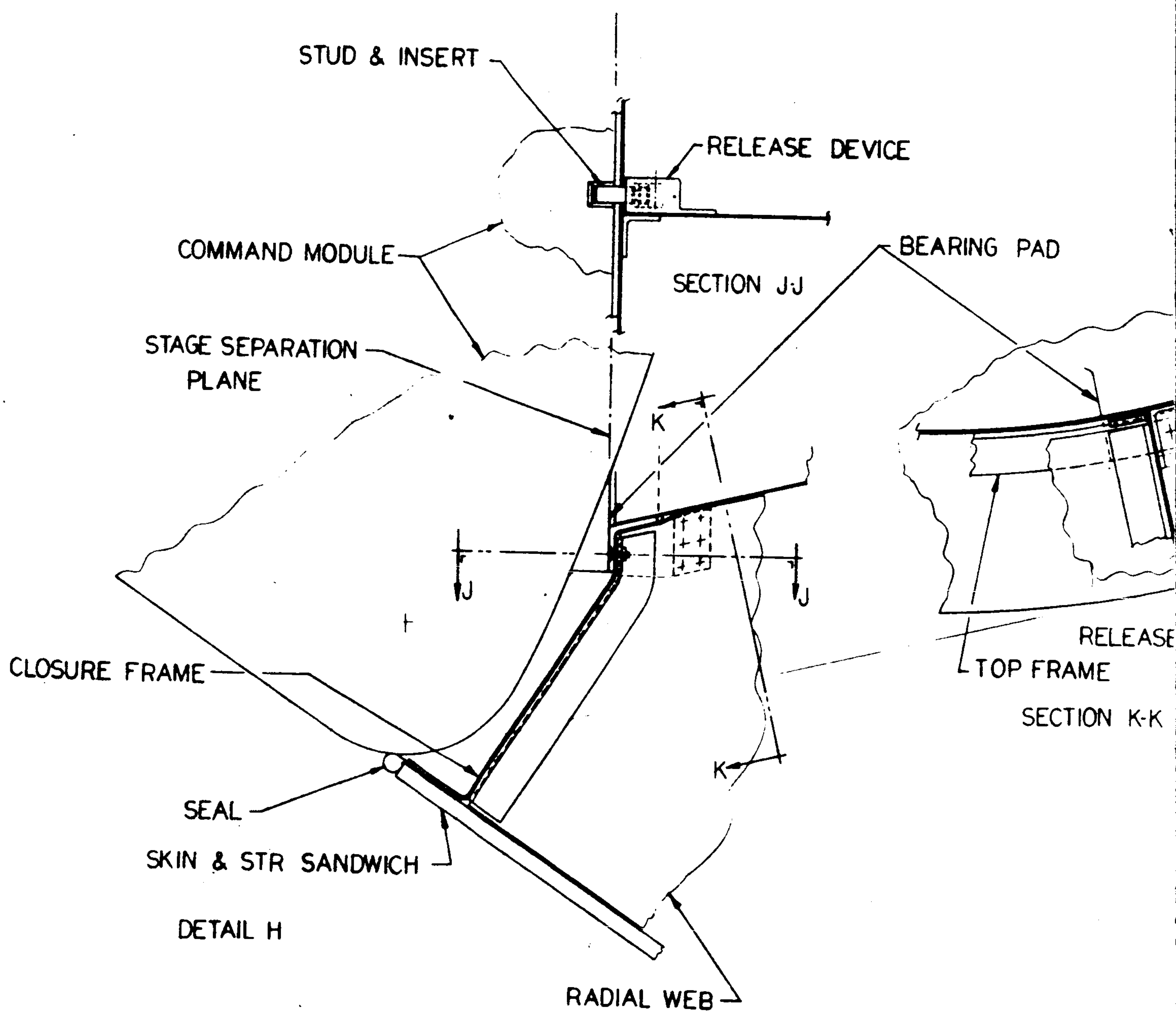
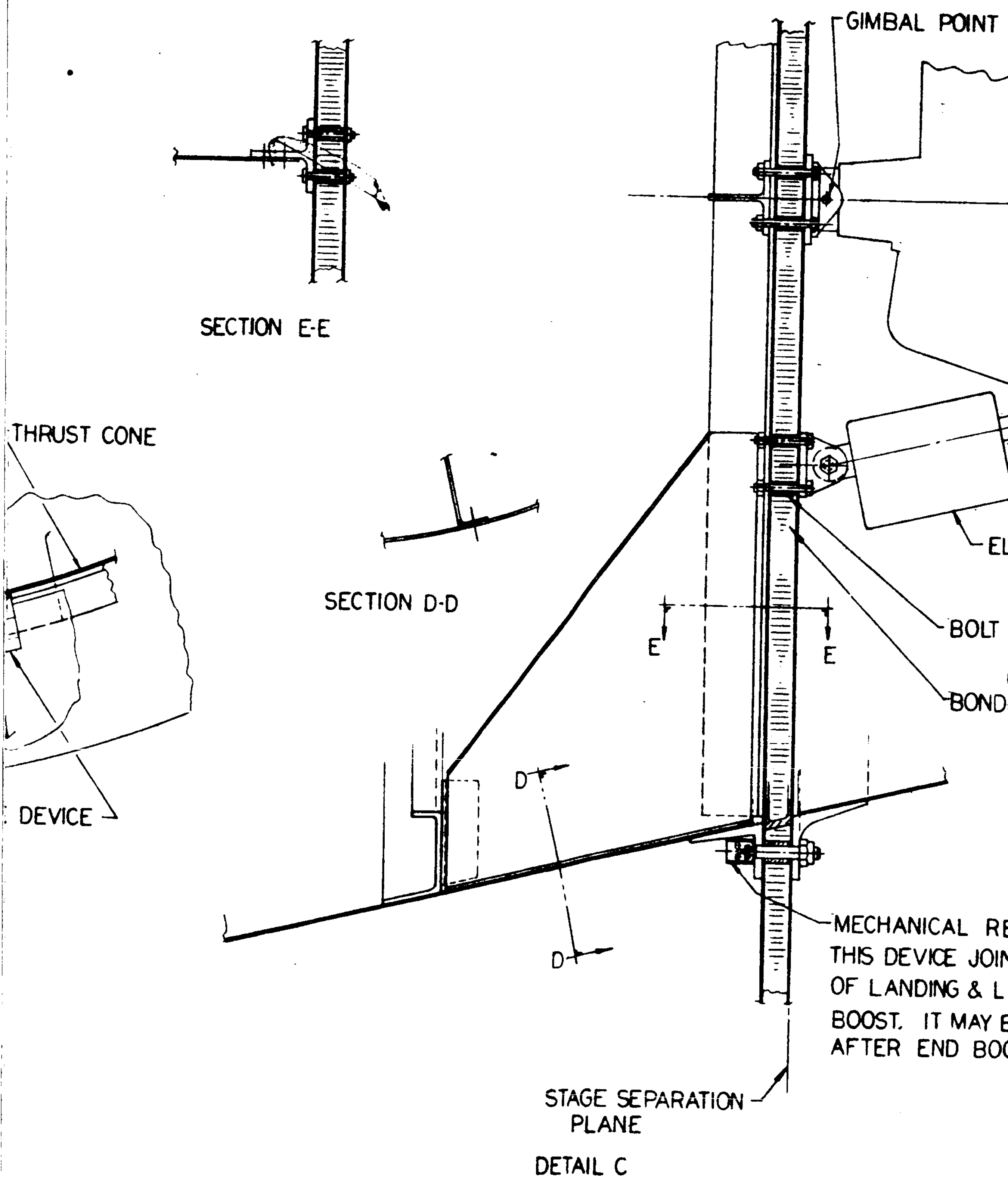
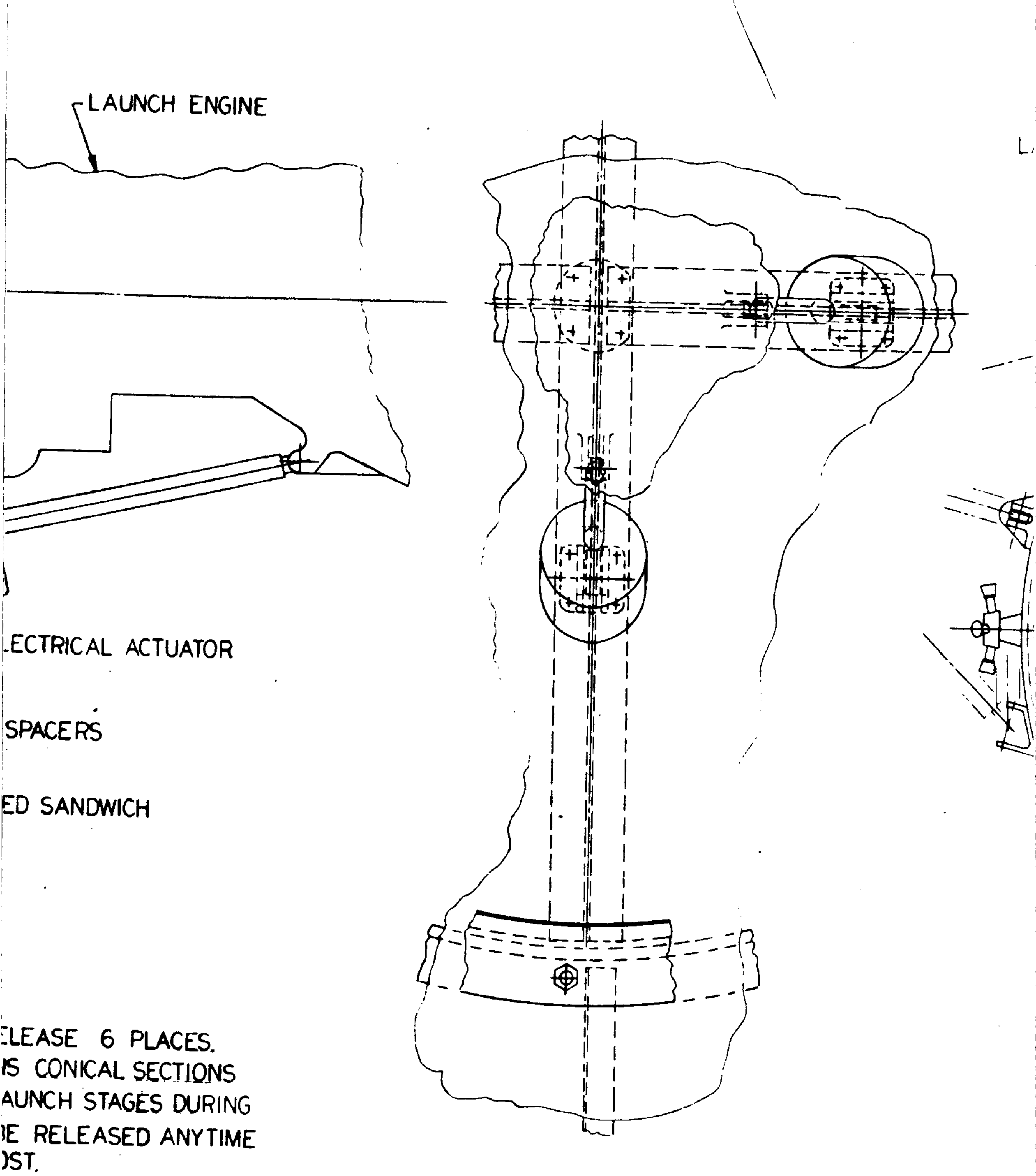
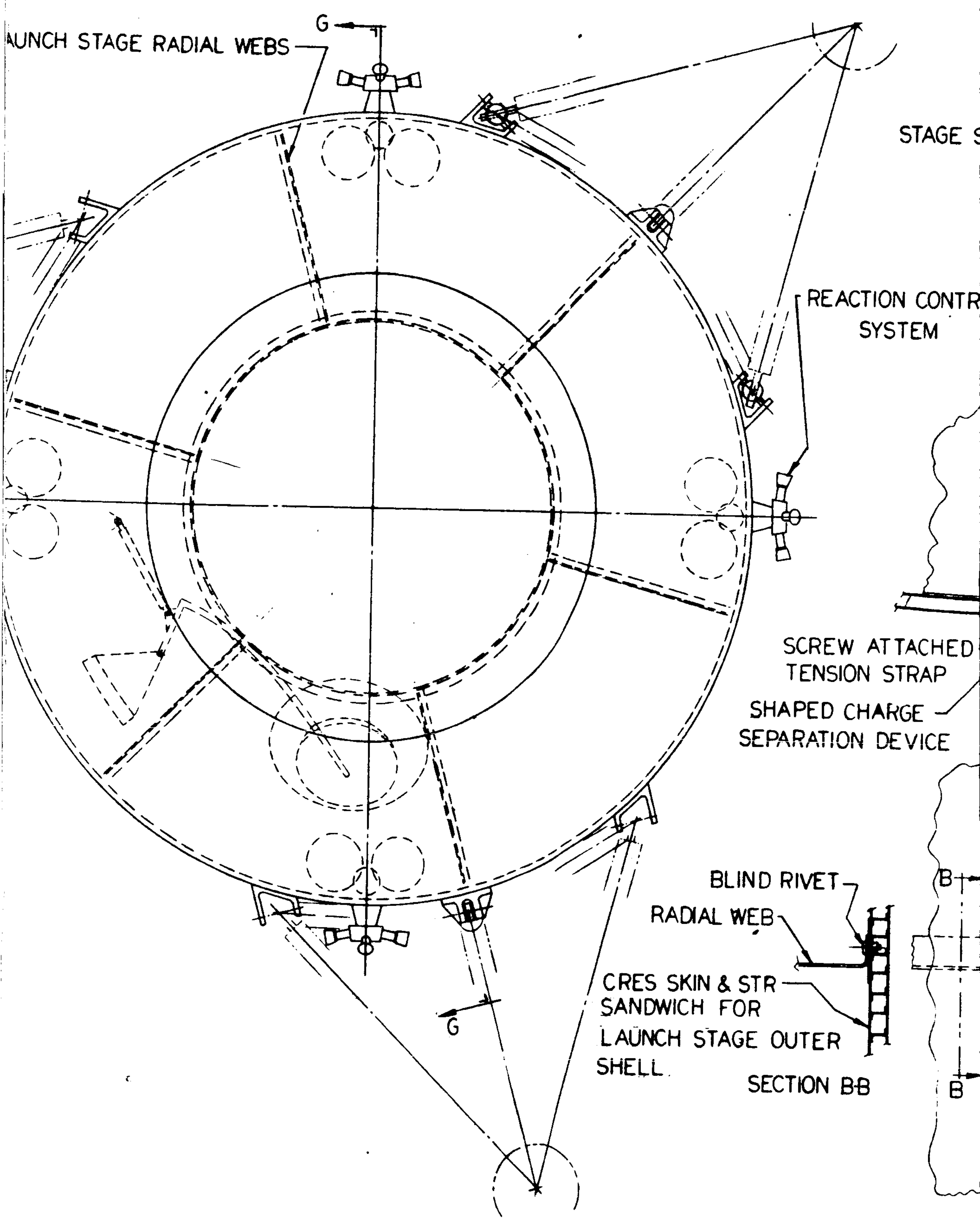


Figure 5.1 Inboard Profile - Lunar Direct Mission Manned Vehicle


STUD & INSERT
RELEASE DEVICE
COMMAND MODULE
BEARING PAD
SECTION J-J
STAGE SEPARATION PLANE
K
J
J
RELEASE
TOP FRAME
SECTION K-K
CLOSURE FRAME
K
SEAL
SKIN & STR SANDWICH
DETAIL H
RADIAL WEB

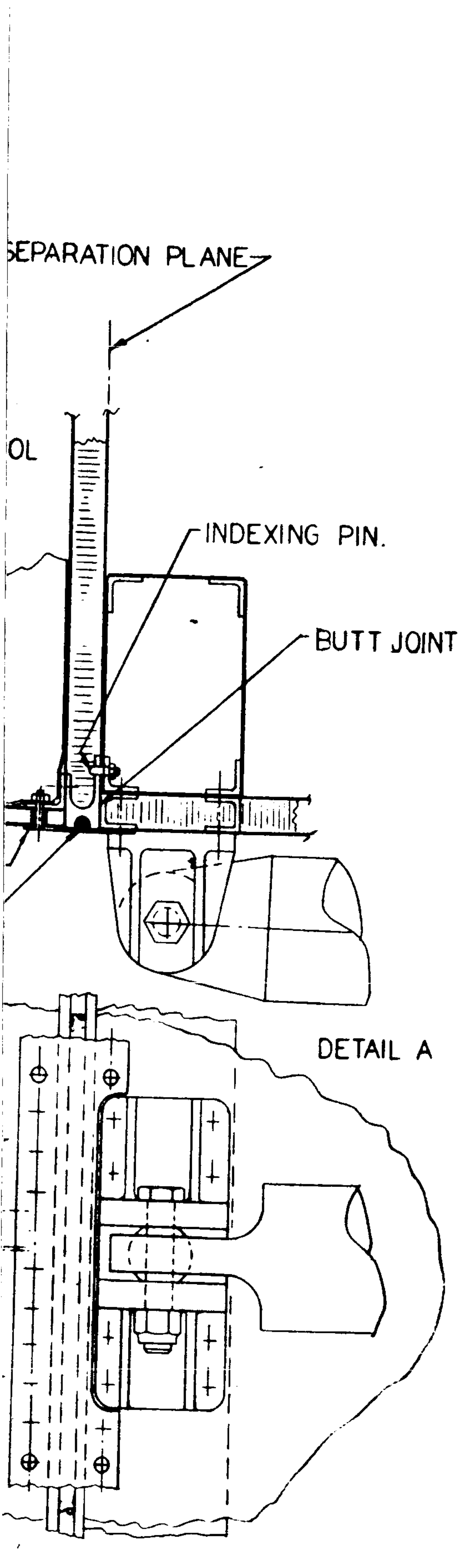

GIMBAL POINT
SECTION E-E
THRUST CONE
SECTION D-D
DEVICE
EL
BOLT
E
E
BOND
D
D
MECHANICAL RE
THIS DEVICE JOIN
OF LANDING & L
BOOST. IT MAY E
AFTER END BOO
STAGE SEPARATION PLANE
DETAIL C

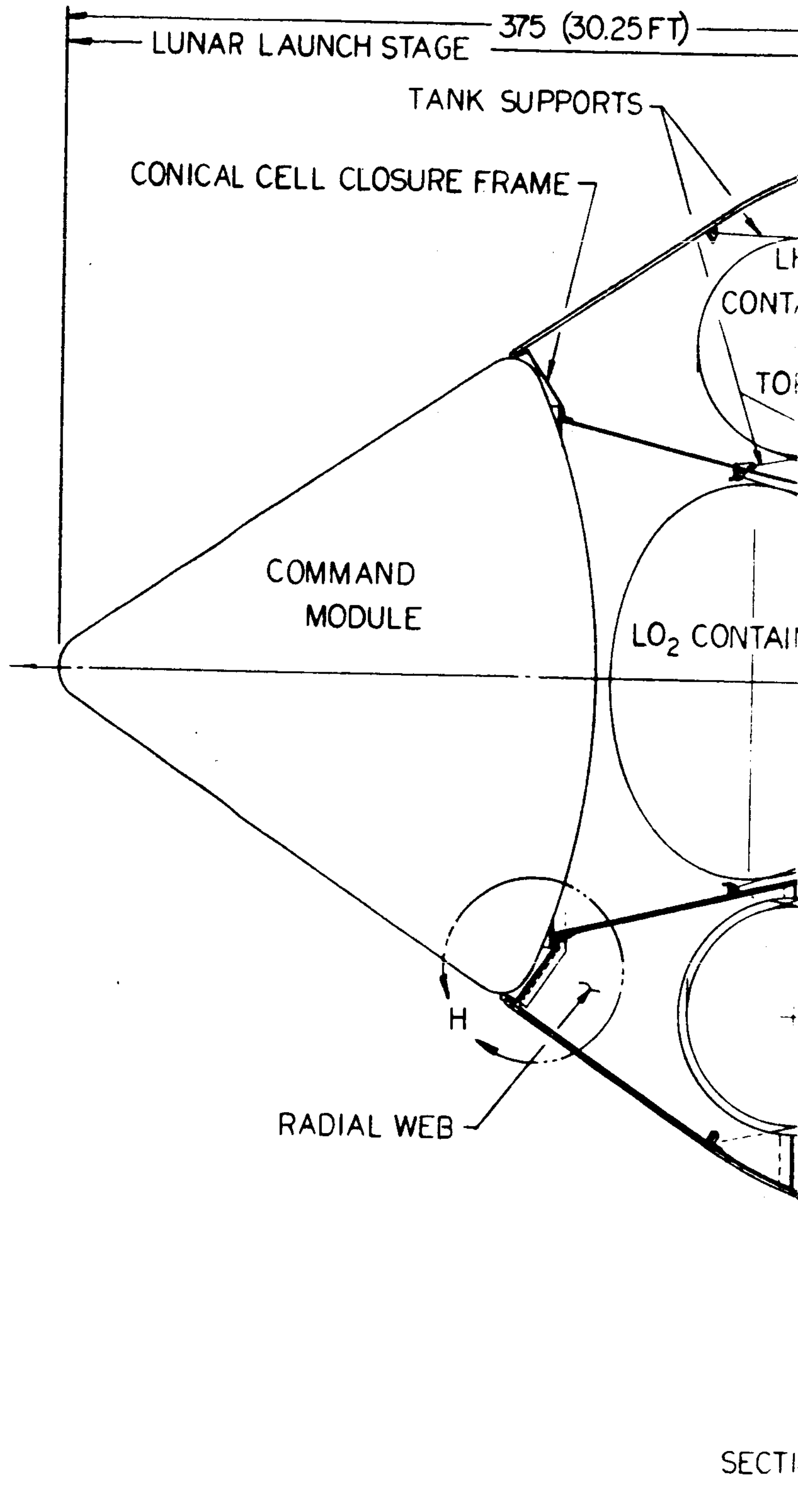

LAUNCH ENGINE
L
LECTRICAL ACTUATOR
SPACERS
ED SANDWICH
LEASE 6 PLACES.
IS CONICAL SECTIONS
AUNCH STAGES DURING
E RELEASED ANYTIME
ST.

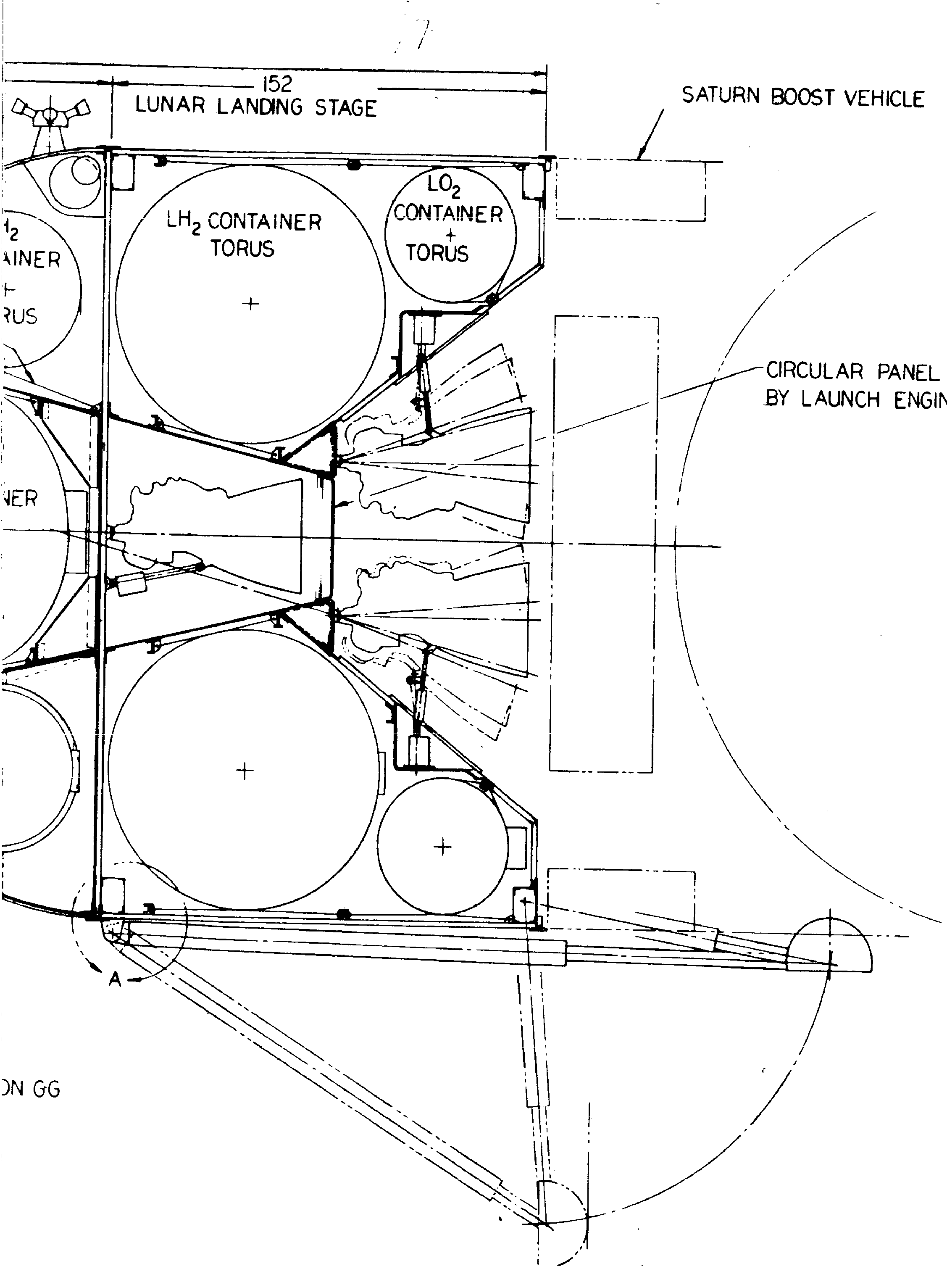


AUNCH STAGE RADIAL WEBS
G
STAGE S
REACTION CONTR
SYSTEM
SCREW ATTACHED
TENSION STRAP
SHAPED CHARGE
SEPARATION DEVICE
B
BLIND RIVET
RADIAL WEB
CRES SKIN & STR
SANDWICH FOR
LAUNCH STAGE OUTER
SHELL
SECTION B-B
G
B

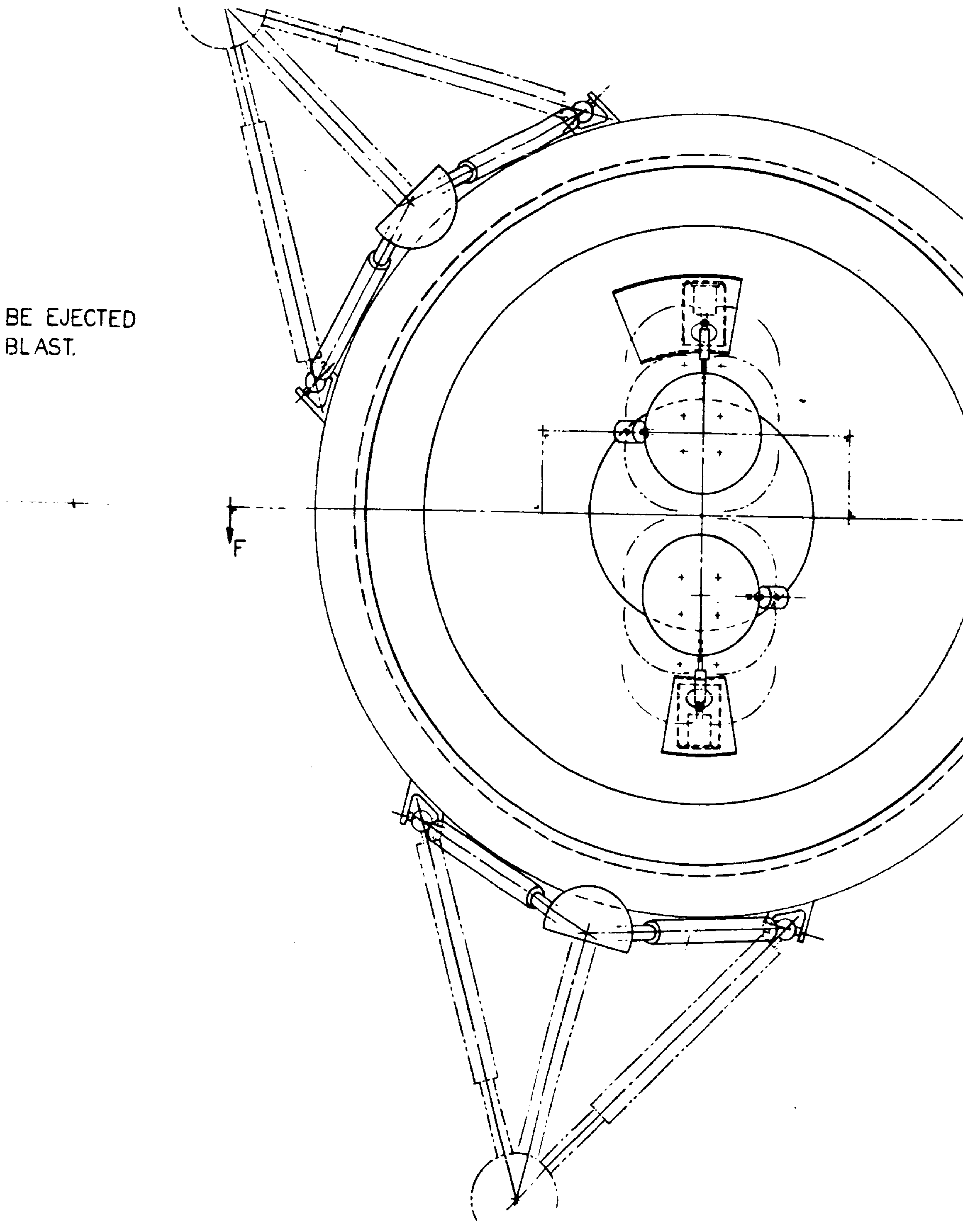
2


SEPARATION PLANE
OL
INDEXING PIN.
BUTT JOINT
DETAIL A


LUNAR LAUNCH STAGE
375 (30.25 FT)
TANK SUPPORTS
CONICAL CELL CLOSURE FRAME
COMMAND MODULE
LO_2 CONTAI
CONT
TO
H
RADIAL WEB
SECTI


152
LUNAR LANDING STAGE
SATURN BOOST VEHICLE
LH_2 CONTAINER TORUS
LO_2 CONTAINER + TORUS
CIRCULAR PANEL
BY LAUNCH ENGIN
A
ON GG

TO BE EJECTED
E BLAST.



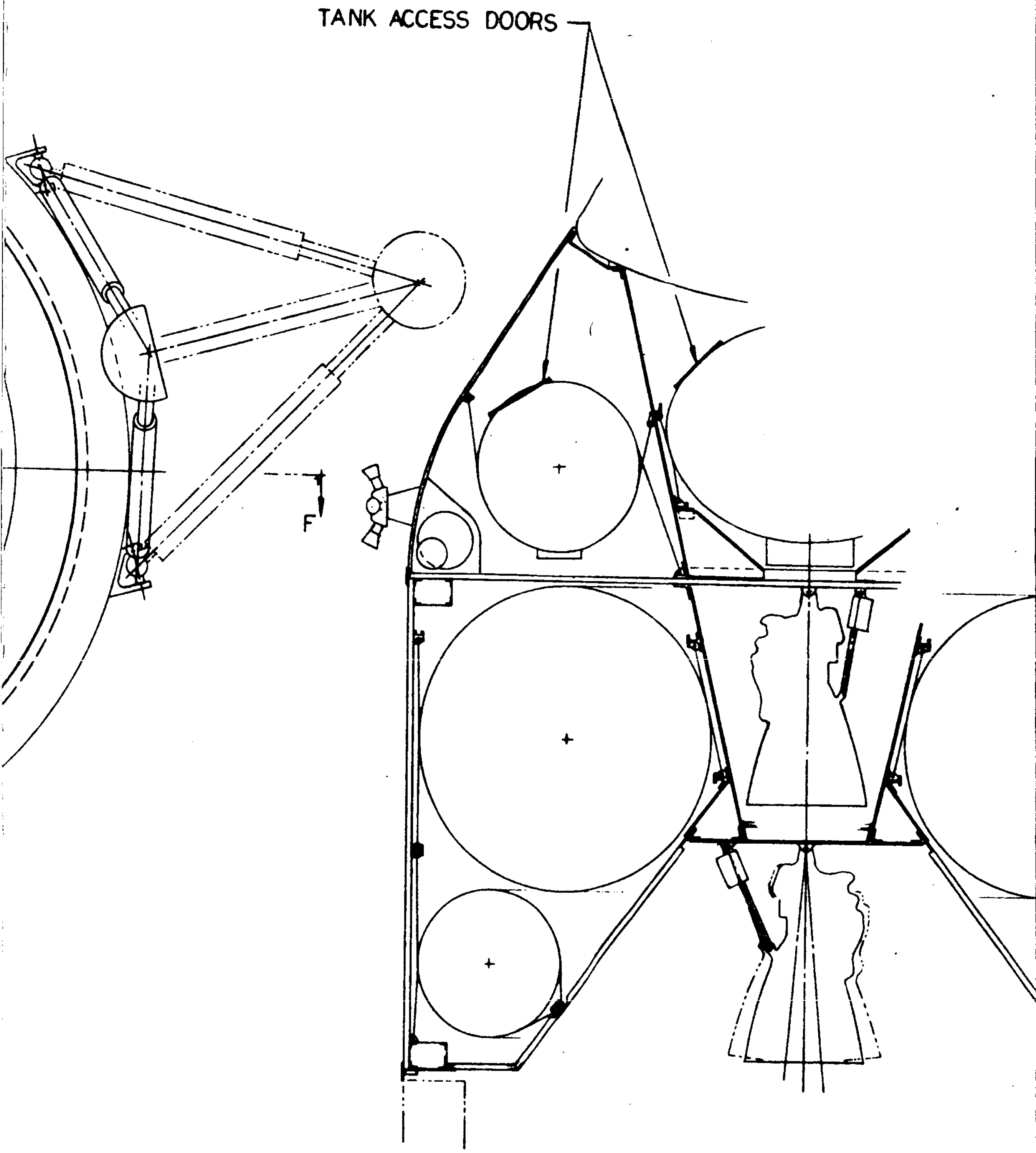
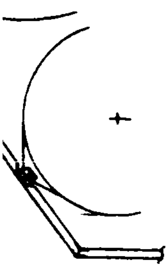


Figure 5.2 S

FOR TYPICAL TANK SUPPORT METHOD
AND LUNAR LANDING STAGE DETAILS,
SEE STRUCTURAL ARRANGEMENT-
LUNAR LOGISTICS BUS.



4

tructural Arrangement - Lunar Direct Mission Manned Vehicle

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[redacted]

by the lunar launch stage and Apollo command module.

5.1.2 Lunar Launch Stage

5.1.2.1 Primary Structure

The primary structure of the lunar launch stage is a corrosion resistant steel skin stringer sandwich, extending from the landing stage separation plane to the Apollo command module. The conical portion of the inner shell acts as the thrust structure for the single RL-10 launch engine.

The inner and outer shell contain circumferential frames which provide mounting points for the multiple cable propellant tank supports. Attachment to the command module will be similar to the present attachment of the Apollo command and service modules. Structural provisions will be made for the attachment of the four reaction control system packages.

5.1.2.2 Propulsion Systems

The lunar launch stage main propulsion system will consist of one fully gimbaled Pratt and Whitney RL-10A-3 pump-fed engine. The reaction control system consists of four modular packages, complete with storable propellant containers, helium pressurant and plumbing. The removable RCS packages are mounted one in each quadrant of the launch stage.

The propellant tanks designed for minimum height and weight of the vehicle are a torus for the liquid hydrogen and an elipsoidal tank for the liquid oxygen. Both tanks will have still wells for liquid level sensing and baffles to dampen propellant sloshing. The liquid hydrogen tank will have a sump compartment equal to approximately 10% of the total tank volume. Internal lines and electrically driven transfer pumps will be mounted in the tank to assure propellant flow to the sump compartment.

The propellant tanks are covered with super insulation and supported with multiple suspension cables to minimize container heat gain. S&ID is currently engaged in a program to study methods and techniques for cryogenic container insulation and support design.

5.1.2.3 Guidance, Navigation and Communication Systems

The guidance, navigation and communication system components are located in the command module. A 54 inch dish antenna is provided in the lunar launch stage for deep space communication. Structural openings for antenna deployment are made with a line shaped charge. A "windowshade" type flexible door covers the opening for meteorite protection when the antenna is deployed. Besides the 54 inch dish antenna, the communication system requires a flush mounted beacon trasponder on the landing stage, a VHF broadband discone antenna on the command module, and four flush mounted C-band antennas on the lunar launch stage.

5.1.2.4 Power and Environmental Control Systems

The fuel cell power and the environmental control systems are similar to the present Apollo design. Space radiators are required for heat dissipation from the ECS system and the fuel cells. These radiators are externally mounted on the outer surface of the lunar launch stage shell. They are hinged at the top and actuated to travel through 90° of arc to provide solar orientation. Consistent with the present Apollo design philosophy, no meteoroid protection is provided for the radiators.

Investigation of high velocity meteoroid impact has been the subject of a laboratory effort at SID but this remains a problem area pending more accurate definition of meteoroid size and frequency in space and on the lunar surface.

Theoretical studies have defined the problem of nuclear radiation from the Van Allen belts, solar flares and other sources but the technique for shielding the vehicle from this radiation is not extensively developed.

5.2 LANDING GEAR

The gear consists of three hemispherical feet, each supported by three struts. Centers of the feet lie on a 230 inch radius. Two of the struts attaching to each foot will be nearly horizontal and will subtend an angle of 60° in a plan view. These struts will attach to the lower box frame. The third strut will extend diagonally upward to the top box frame and lie midway between the two lower struts.

During boost, the struts will lie against the outer surface of the vehicle extending below the boost vehicle separation plane. The hemispherical feet will be allowed to penetrate the boost vehicle skirt in a manner which will not interfere with its equipment package. The gear will be extended a short time prior to separation from the boost vehicle so as not to interfere with separation.

All attachments of the struts to the feet and to the structure will be by universal joints. The feet will contain a crushable material such as aluminum honeycomb which will resist springback. The struts have been visualized as resembling hydraulic cylinders, but will contain the crushable material rather than a fluid. Pistons inside the struts will make butt contact with the crushable material when the gear is fully extended. On landing impact the piston will be forced against the crushable material, shortening the strut by a predictable dimension, which will be determined by tests.

A detail study of landing gear requirements and laboratory testing of a model is currently active at the SID Development Laboratory. It is felt that the detail landing gear design should reflect the results of this investigation which is not yet completed.

5.3 INTERFACE DESIGN

The areas of internal interface include the command module/lunar launch stage attachment, the lunar launch stage/lunar landing stage attachment and the landing stage/boost vehicle attachment. The areas of external interface include the spacecraft to ground service umbilicals and propellant service lines. In space and on the lunar surface, a communications interface exists.

Matching of mechanical and structural components, interstage plumbing and umbilical disconnects, release mechanisms and bolted fittings will require master tooling fixtures and mock-ups to simulate adjacent parts and structure. Ground service connections will be compatible with prelaunch checkout equipment and procedures. Personnel access for checkout and service repair has been considered in the configuration design. No in-flight access to the stages has been provided.

The separation system for the command module from the lunar launch stage is similar to the present mechanical release provided for the Apollo Command Module. Interstage separation between the Lunar Launch and Landing stages and between the landing and boost vehicle stages is obtained with line shaped charges imbedded under the sandwich skin. Bolted rings at these locations provide a means of stage assembly. Master tooling is required to coordinate the mating surfaces.

REFERENCES:

1. NAA-S&ID Report SID 62-1189, "Lunar Mission System Studies - Interim Report," 30 October 1962

2. NAA-S&ID Report SID 62-1466, "Lunar Logistic Vehicle Study," 31 December 1962

6.0 PROPULSION

Propulsion is provided for both the landing and take-off stages of the Direct Manned Lunar Landing Vehicle. These systems were designed in sufficient detail to assure proper integration with the vehicle, and to allow vehicle design, performance and weights to be determined. The landing stage propulsion system of the Direct Manned Lunar Landing Vehicle is identical to the landing stage propulsion of the Lunar Logistic Vehicle (Reference 1), except for the increased propellant quantity in the reaction control system. The primary subject of this report is propulsion for the take-off stage, although the landing stage propulsion is summarized for reference purposes. The propulsion system concept and major components are identical for both stages.

The take-off stage propulsion system provides thrust for accomplishing the lunar take-off/orbit injection and trans-earth injection phases of the mission. The reaction control system provides attitude and translational control of the vehicle.

During the course of the study, various engine arrangements and propellant tank configurations were analyzed for a variety of vehicle designs and staging concepts. Also, an extensive analysis was made of various pressurization methods. The results of these studies were repprted in References (2) and (3). The final selected system is the primary subject of this report.

6.1 GENERAL DESCRIPTION

Main take-off propulsion is provided by a 15,000 thrust, pump-fed engine which is identical to the landing engines except that it is operated at a fixed thrust level as a non-throttled engine. The engine is gimbal-actuated to provide for thrust vector control. The liquid oxygen and liquid hydrogen propellants are fed to the engine from an ellipsoidal-shaped oxidizer tank and from a sump compartment within the toroidal-shaped fuel tank. Transfer pumps, located in the fuel tank, transfer the propellant to the sump compartment. Pressurization of the propellant tanks is provided by a supply of helium, stored at 4000 psi in a pressure vessel that is submerged in the liquid oxygen. During engine operation, the fuel tank is pressurized by gaseous hydrogen bled from the engine.

Reaction control propulsion is provided by sixteen 100 lb thrust pressure-fed thrust chambers which are pulse-modulated for precise control. The thrust chambers are grouped in four radial clusters of four chambers each to provide pitch, yaw, roll and translational control, and includes redundant chambers to account for Reaction Control System (RCS) engine-out failures. The storable-type propellants, pressurized by helium, are divided into four independent propellant feed systems, with one system located at each of the four chamber clusters.

6.2 ENGINE SYSTEM

The take-off engine is an advanced version of the Pratt and Whitney RL 10A-3 rocket engine presently used in the Centaur and Saturn S-IV vehicles. The advanced version of the RL10 includes improvements and increased capability which are presently being funded by NASA.

The major engine requirements for the take-off stage are:

1. Fixed thrust level
2. Multi-start capability; a minimum of two starts
3. Instant start capability
4. Gimbal capability of $\pm 5^{\circ}$

6.2.1 Engine Description

A simplified engine schematic drawing is shown in Figure 6.1. The major components of the engine are a thrust chamber, turbopump assembly, and control valves. The thrust chamber is of tube-wall construction and is regeneratively cooled with the hydrogen fuel. The entire length of the high expansion ratio nozzle is cooled with hydrogen. Hydrogen gas which leaves the thrust chamber cooling jacket is expanded through a turbine which drives the oxidizer and fuel pumps. After passing through the turbine, hydrogen gas is then injected into the chamber to burn with the liquid oxygen. Ignition in the combustion chamber is provided by a spark ignition system. Thrust control is obtained by regulating the flow of hydrogen bypassed around the turbine, and by regulating the flow of oxygen. For the take-off stage, the engine thrust control maintains constant thrust. (The throttable RL-10 used in the landing stage has automatic control of the thrust level provided by inputs from the vehicle flight control system and by movement of the pilot's throttle). Mixture ratio control is provided by inputs from the propellant utilization system which is described under "Propellant Feed System".

Additional information on the advanced RL10 engine used in this

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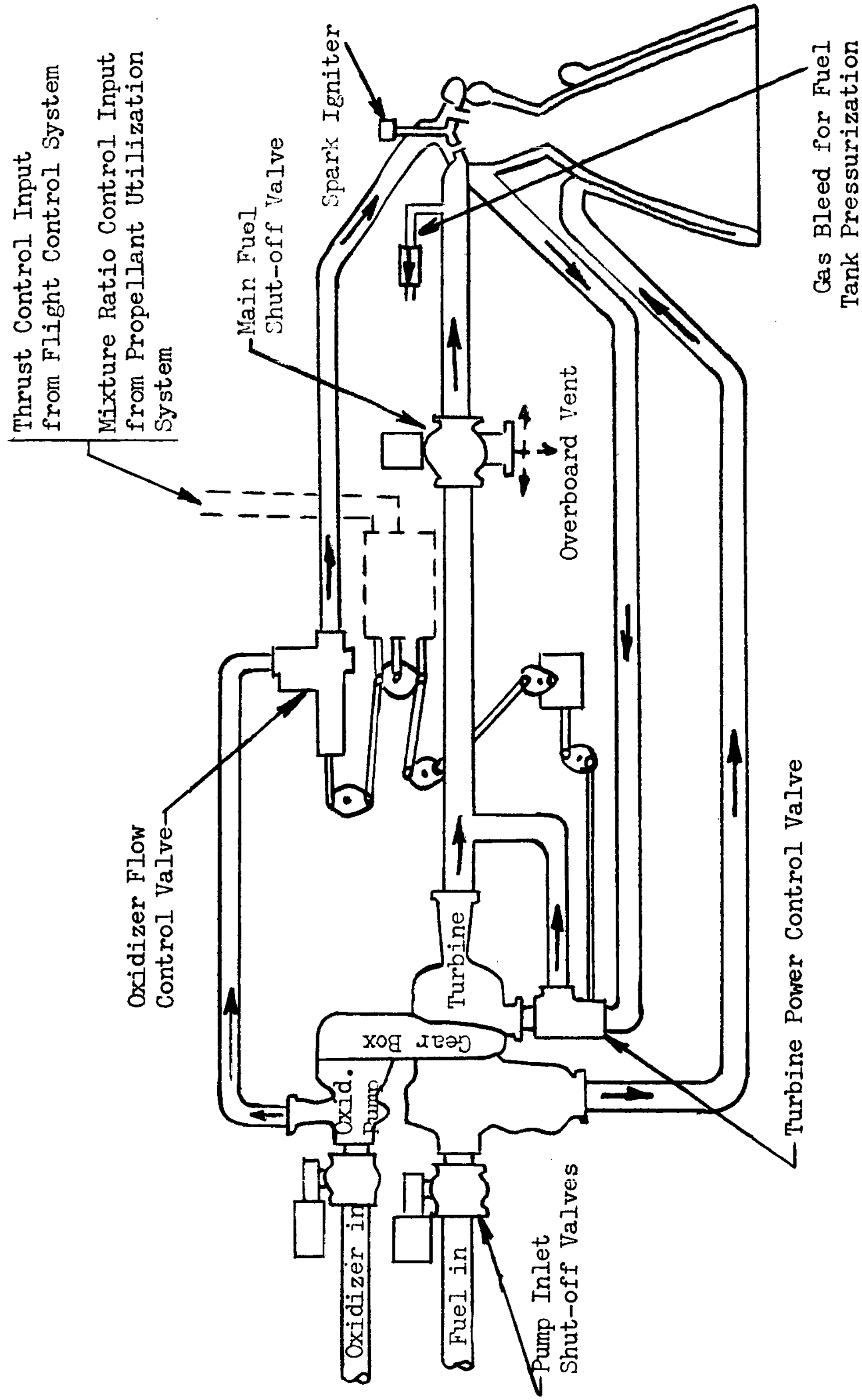

FIGURE 6.1

ADVANCED RL-10 ENGINE FLOW SCHEMATIC

[redacted]

study may be found in the interim reports of References (2) and (3) and in the engine manufacturer's reports of References (4) and (5), which give more detailed data. Reference (6) provides the engine manufacturer's plan for conducting a throttling feasibility program to demonstrate engine throttling of the RL-10 to 10% of rated thrust which is required for the landing stage engines. Design and operating details of the present RL-10A-3 engine were obtained from References (7) through (10).

The take-off engine is installed in the vehicle on gimbaled thrust mounts. Refer to Figure 5.2. Electromechanical gimbal actuators provide for pitch and yaw control to ± 5°. The electromechanical actuators were selected to provide for reliable operation during the long exposure to space vacuum and to avoid leakage problems with hydraulic and pneumatic systems.

The engine installation includes a supply of helium for operation of engine components, and is duscussed in further detail in the propulsion section under "Propellant Feed and Pressurization Systems."

6.2.2 Engine Operation

The take-off engine buring durations for each phase of the mission profile is shown in Table 6-II. The propellants were allotted to each burning phase of the mission according to the breakdown of ΔV's given in Section 4, Mission Analysis, of this report.

The vacuum performance of the advanced Pratt & Whitney RL-10 engine used in the take-off stage is shown in Table 6-I.

The trans-Earth mid-course correction is accomplished with the reaction control system.

In addition to normal pump-fed, the engine may be operated in the ullage thrust mode on vaporized propellants. Consideration was given to

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TABLE 6-I

RL-10 VACUUM PERFORMANCE

Thrust	15,000 lb
Minimum Specific Impulse	425 sec
Propellants	LO_2/LH_2
Mixture Ratio, O/F	5 to 1
Chamber Pressure	300 psia
Nozzle Expansion Ratio	40 to 1
Engine Weight, Dry	295 lb
Engine Weight, Wet	302 lb
Total Propellant Flow Rate	35.3 lb/sec
Oxidizer Flow Rate	29.4 lb/sec
Fuel Flow Rate	5.9 lb/sec
Oxidizer Pump NPSH	15 psi
Fuel Pump NPSH	8 psi

TABLE 6-II

ENGINE BURNING DURATION

MISSION PHASE	NO. OF FIRINGS	THRUST LB	T/W	BURNING DURATION SEC
1. Launch & Lunar Orbit Injection	1	15,000	0.533	289
2. Trans-Earth Injection	1	15,000	0.835	112
		TOTAL BURNING DURATION:		401

TABLE 6-III

ULLAGE THRUST MODE PERFORMANCE

Oxidizer pressure / Fuel pressure	65 psia / 44 psia			45 psia / 30 psia		
Percent Gaseous Oxidizer	Fn Lb	Pc psia	I_{sp} sec	Fn Lb	P_c psia	I_{sp} sec
20	780	15	388	530	10	382
40	680	13	400	450	8.5	398
100	540	10.4	385	350	7	383

this mode of operation as the initial part of a normal start to clear the lines of any gasified propellants. In this pressure-fed mode, the engine operates at low chamber pressure and propellant flow rates. The propellants are supplied by the ullage pressure in the propellant tanks. The engine system includes valving to allow the fuel to bypass the pump turbine and the oxidizer to bypass the oxidizer control valve. The performance of the engine in the ullage thrust mode is shown in Table 6-III for two different combinations of propellant inlet pressure conditions and for different percentages of gasified oxidizer. Fuel is injected in the gaseous state for either the ullage or pump-fed modes.

In summation, it should be noted that the major difference in the engine system described herein and that reported previously in References (2) and (3) is that two engines are now used for landing instead of various firing sequences with three engines that included the take-off engine. This two-engine landing mode has decreased the burning duration of the take-off engine somewhat and reduced the minimum thrust level required for landing, which, in the previous study phase, was accomplished by the single take-off engine.

This arrangement has also resulted in a simplified feed system, since it is no longer required to feed the take-off engine from both the landing as well as take-off stages as was the case previously.

6.3 PROPELLANT AND PRESSURIZATION SYSTEM

The function of the propellant and pressurization system is to deliver liquid oxygen and liquid hydrogen to the RL-10 engines within the pressure and temperature limitations established for the RL-10 and to make maximum utilization of the propellant on board. The take-off stage propellant and pressurization system consists of the following sub-systems:

1. Propellant feed
2. Internal tank configuration and transfer system
3. Vent and relief
4. Propellant management
5. Pressurization

The propellant flow rate required by the RL-10 engine is 29.4 lb/sec LO_2 and 5.9 lb/sec LH_2.

Figure 6.2 shows the LO_2 and LH_2 pressure and temperature envelope requirements for the RL-10A-3, Reference (7). These envelopes were used as a representative requirement for an advanced RL-10 engine considered for this study. Pratt & Whitney has stated that the minimum and maximum pressures shown in Figure 6.2 for the pre-start operation region do not represent an engine capability limit but were defined by programs presently using the RL-10. It should also be noted that the pressure requirement for starting an engine is somewhat greater than for steady state engine operation.

The engine also imposes a requirement that each propellant feed line have a flexible section to allow for engine gimbaling. The gimbaling requirement is approximately $\pm$ 5° in both the pitch and yaw planes.

One engine start is required to be made while the vehicle is in an apparent zero gravity environment. The other engine start is made prior to take-off from the lunar surface.

Maximum utilization of the propellants is required to maintain vehicle weight at a minimum.

6.3.1 Propellant System

A schematic of the propellant system is shown in Figures 6.3 and

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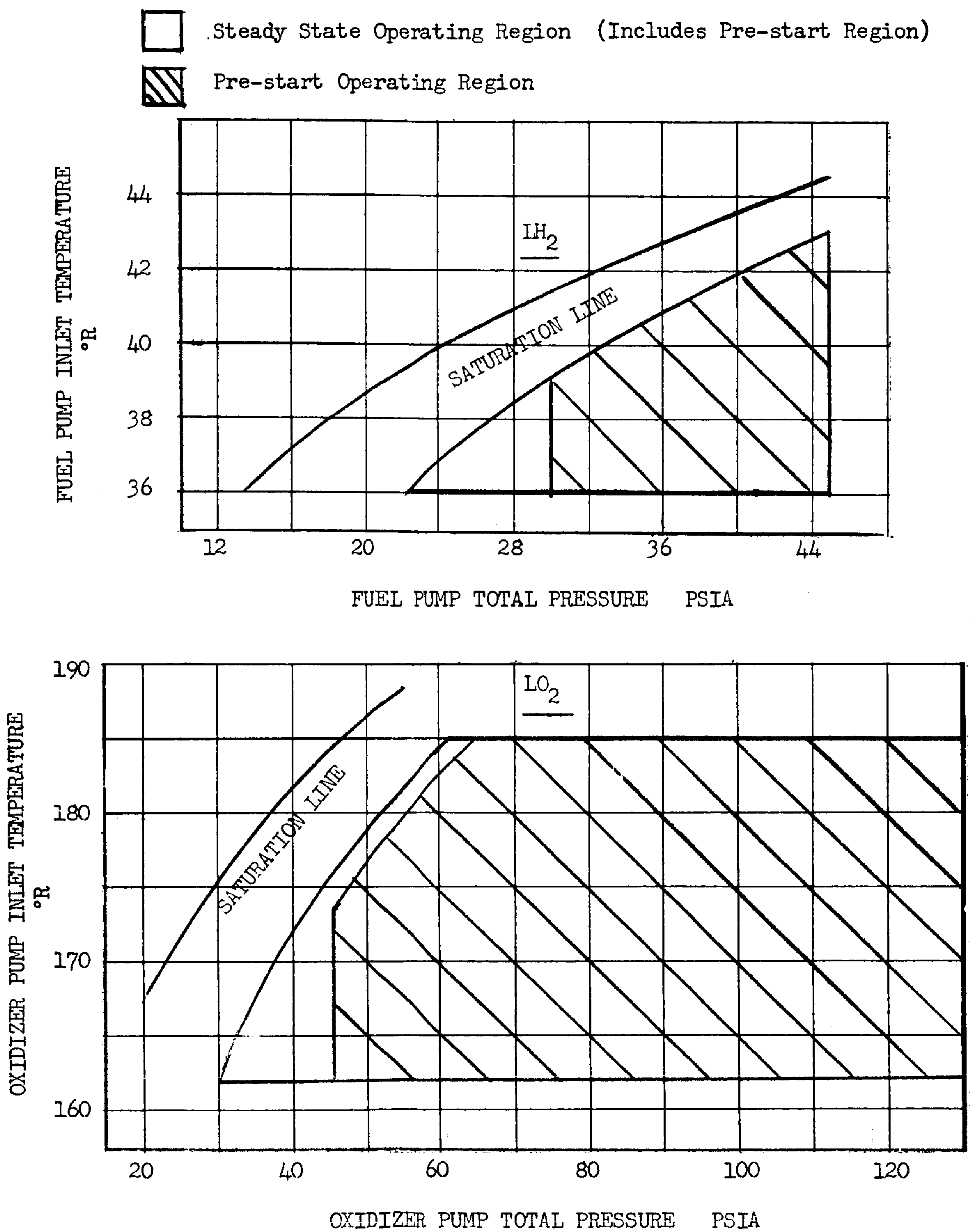

FIGURE 6.2

PROPELLANT CONDITIONS REQUIRED AT RL-10 PROPELLANT PUMP INLETS

6.4. Figure 6.3 depicts the propellant flow path and valving concept, while Figure 6.4 depicts the internal tank configuration and LH_2 transfer system. The LH_2 tank is toroidal shaped and the LO_2 tank is ellipsoidal shaped. These tank shapes allow the most compact packaging and lightest vehicle total weight, Reference (2). Propellant flow to the engine is controlled by the liquid oxygen and liquid hydrogen pump inlet shut-off valves, which are part of the RL-10 engine and are mounted directly on the turbopump inlets. These shut-off valves are helium actuated, normally closed, ball-type with spring-loaded resilient plastic seals to minimize propellant leakage. The system valving concept shown here is simplified from the system shown in the interim report, Reference (2), by the elimination of prevalves from the vehicle side of the interface. The propellant lines will be full of propellant at all times. Consideration has been given to isolating the propellants within the propellant tanks to minimize heat leak into the propellants. However, the LO_2 line is less than two feet long and does not warrant the added valve. The valve was also eliminated from the LH_2 line for system simplicity. If isolation of propellants within the tanks were shown to be of great advantage in reducing heat leak into the propellants, squib-actuated valves would be added which are opened prior to lunar take-off. These valves would remain open for the remainder of the mission.

Flexible bellows are provided in each line to allow for engine gimbaling of approximately $\pm$ 5° in both the pitch and yaw planes.

6.3.2 Fill and Drain

A fill and drain nozzle and isolation valve are provided. The fill and drain nozzle incorporates a shut-off feature, but an isolation valve is also provided to decrease the possibility of propellant leakage.

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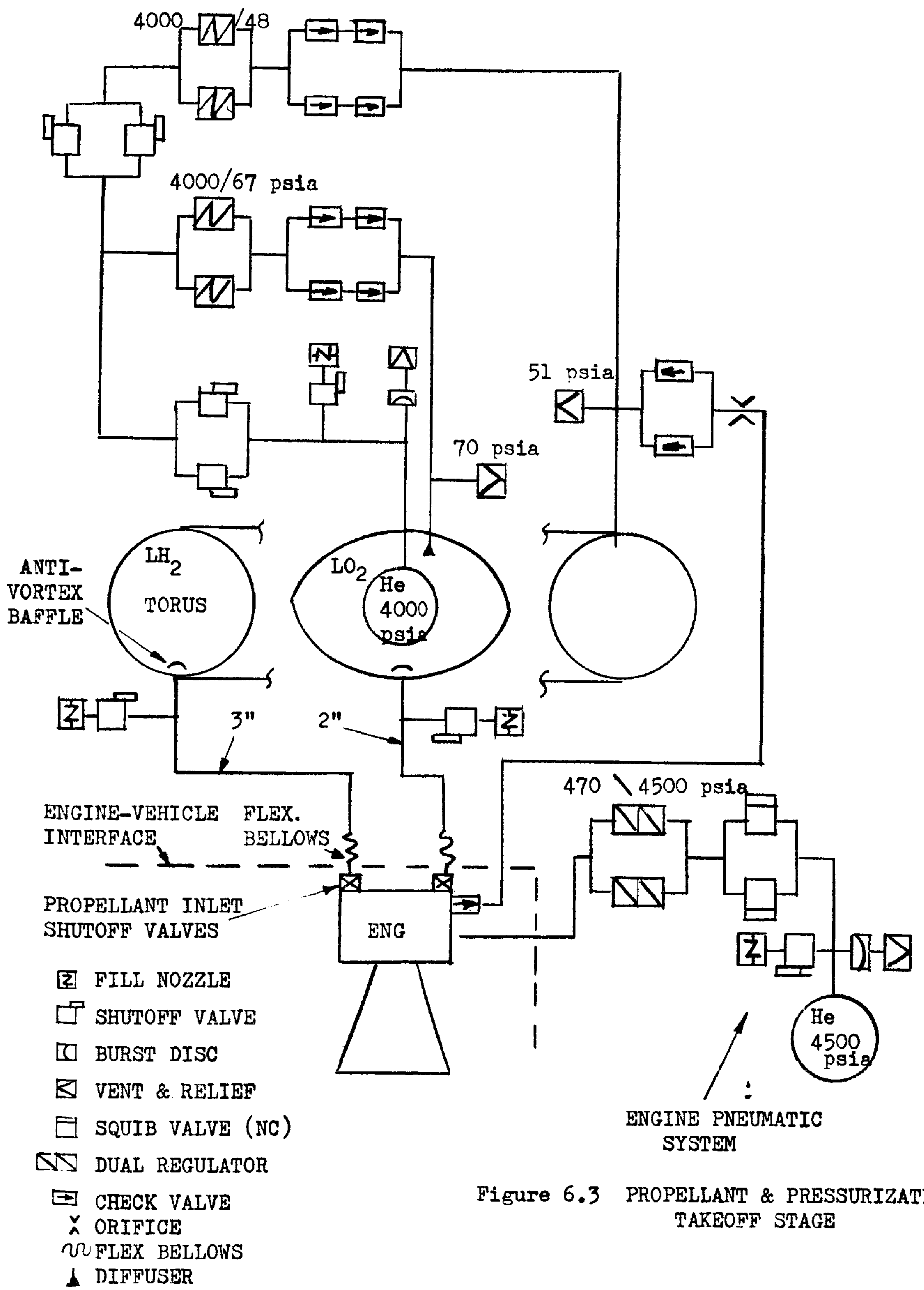

Figure 6.3 PROPELLANT & PRESSURIZATION, TAKEOFF STAGE

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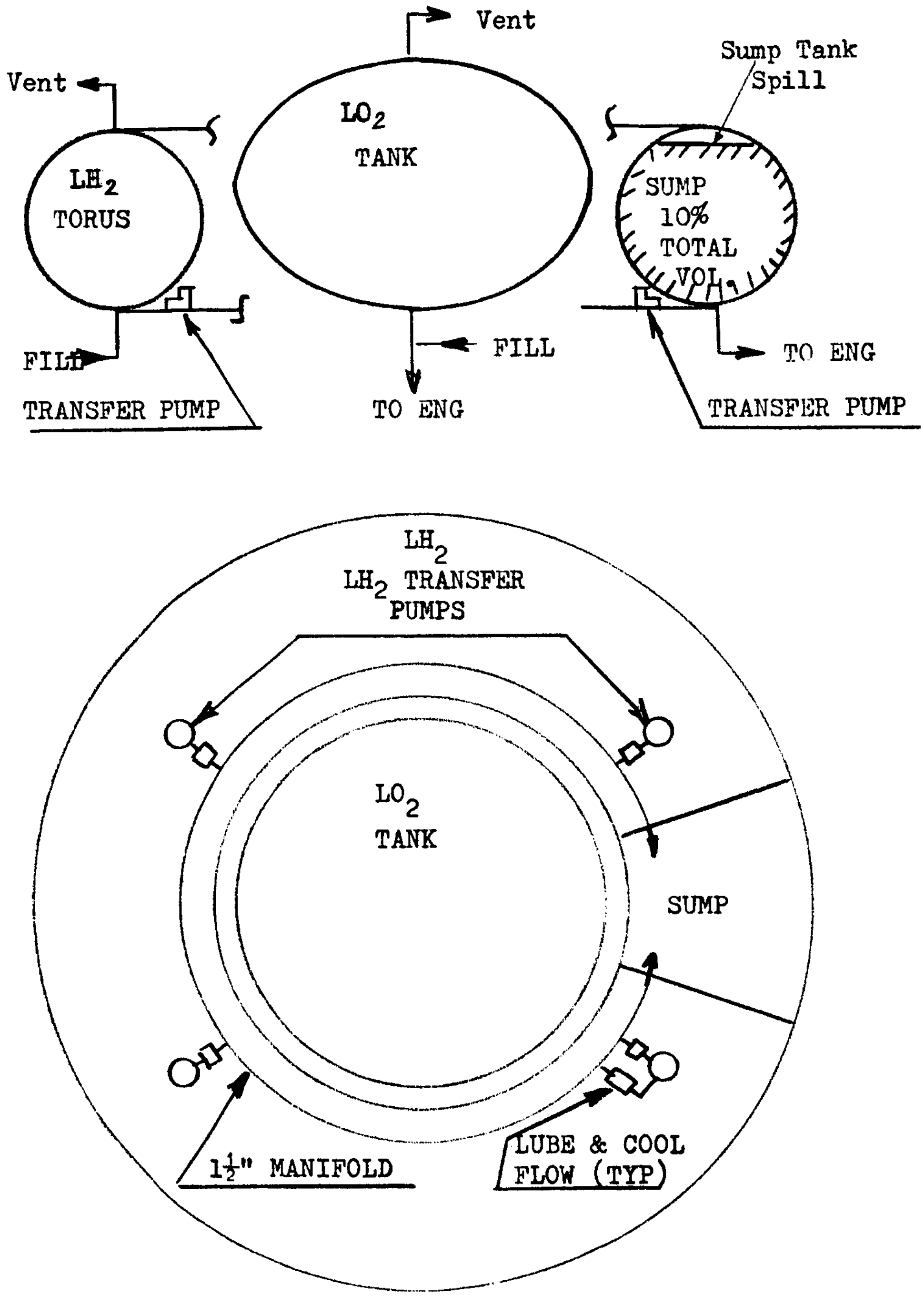

Figure 6.4

INTERNAL TANK CONFIGURATION & TRANSFER PUMP SYSTEM

6.3.3 Vent System

A vent and relief system is provided to protect the propellant tanks from over-pressurization and to allow tank venting during propellant servicing. The vent system pressure relief is set at 70 ± 1 psia and 51 ± 1 psia for the LO_2 tank and LH_2 tank respectively. The exact type of system has not been determined. During main engine operation and lunar stay, any gas within the propellant tank will be at the top of the tank, so a simple vent system using a relief valve located at the top of the tank can be used. However, during the zero gravity coast periods, a zero gravity vent system similar to the one to be used on Centaur may be required. The Centaur system uses a centrifugal separator, turbine, and heat exchanger. Centrifugal force is used to separate the gas and liquid of a two-phase fluid. The liquid is returned to the propellant tank and the gas is passed through a turbine. The turbine extracts energy from the gas, thereby cooling the gas, to drive the centrifugal separator. Then the cooled gas is passed through a heat exchanger to absorb heat from the main propellant tank before being dumped overboard. This system is obviously more complex than normal vent systems and has not been proven in zero gravity flight, but this type of vent system may be required to prevent dumping large quantities of liquid propellant overboard. Since the Centaur type system is designed to separate a two-phase fluid (gas and liquid), the vent system inlet must be located where the fluid is always a mixture of gas and liquid, or all gas.

6.3.4 Propellant Management

A propellant management system is included to provide for automatic and manual propellant loading and for propellant utilization during engine operation. The function of the propellant management system is to minimize the residual propellant and, therefore, reduce the total vehicle weight. This is accomplished by ensuring that the proper amount of propellant is on board at launch, then determining the amount of each propellant remaining throughout the mission, and providing commands to vary the RL-10 mixture ratio so the LO_2 and LH_2 are depleted simultaneously. The RL-10 mixture ratio can be varied from 4.4 to 5.6, Reference (7). The use of propellant utilization on past vehicles has resulted in improving the vehicle total impulse and, therefore, increased range or payload capability over the same vehicle flown without a propellant utilization system. A propellant management system will also provide crew confidence and mission safety by efficient use of the propellant and display of remaining propellant to the crew. This display of remaining propellant is especially vital prior to lunar take-off to ensure that sufficient propellant is on board to safely complete the return mission.

The propellant management system concept favored at this time is the point-sensor concept with two integrating mass flowmeters; one in each propellant line. The functional concept of a point-sensor system is that sensors will be located in each propellant tank with each LO_2 sensor having a mate in the LH_2 tank. Both mating sensors should uncover simultaneously if the desired mixture ratio has been maintained. If one sensor uncovers later than the mate, an error signal is detected and the mixture ratio of the engine is changed by command from a computer to correct this error before the next sensor pair uncovers. The point-sensors are located within stillwells to reduce the effects of slosh on the measured quantity of propellant remaining. Several stillwells are located within the LH_2 toroidal tank to reduce the effect of propellant tilt with respect to the vehicle base on the meas-

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ured quantity of propellant remaining. One stillwell is located in the center of the LO_2 ellipsoidal tank. Information from the two integrating mass flowmeters will be used to give the crew continuous readout of the propellant remaining. The point-sensor data, which is more accurate than the flowmeter, will be used to correct the flowmeter readout periodically.

6.3.5 Propellant Feed System

A sump compartment in the LH_2 tank is equal to approximately 10% of the total tank volume. The ellipsoidal LO_2 tank geometry forms a natural sump. The LH_2 sump compartment will contain the final LH_2 used by the main engine for trans-Earth injection. The confinement of the final LH_2 to a sump will minimize the amount of unavailable LH_2 in the toroidal tank. The LH_2 tank has transfer pumps which transfer LH_2 to the sump compartment where it is fed to the engine by a pressurization system. The pumps are manifolded together and the discharge of each pump is checked to prevent propellant back flow through a failed pump. A typical pump installation is shown in Figure 6.5.

Four transfer pumps are located in the LH_2 tank with only three pumps required to provide the maximum engine flow demand. The top of the sump compartment is open to the remainder of the LH_2 tank to allow any excess transfer flow to return. Four pumps were selected as the probable number required to minimize unavailable LH_2 while maintaining the simplest practical system. The pumps will be driven by AC motors which are submerged in the LH_2. Propellant flow is returned from the manifold to provide cooling amd lubrication of a pump in the event that it is uncovered. As discussed in more detail in Reference (3), operation at cryogenic temperature allows the motors to be lightweight and have high efficiency. The electrical power

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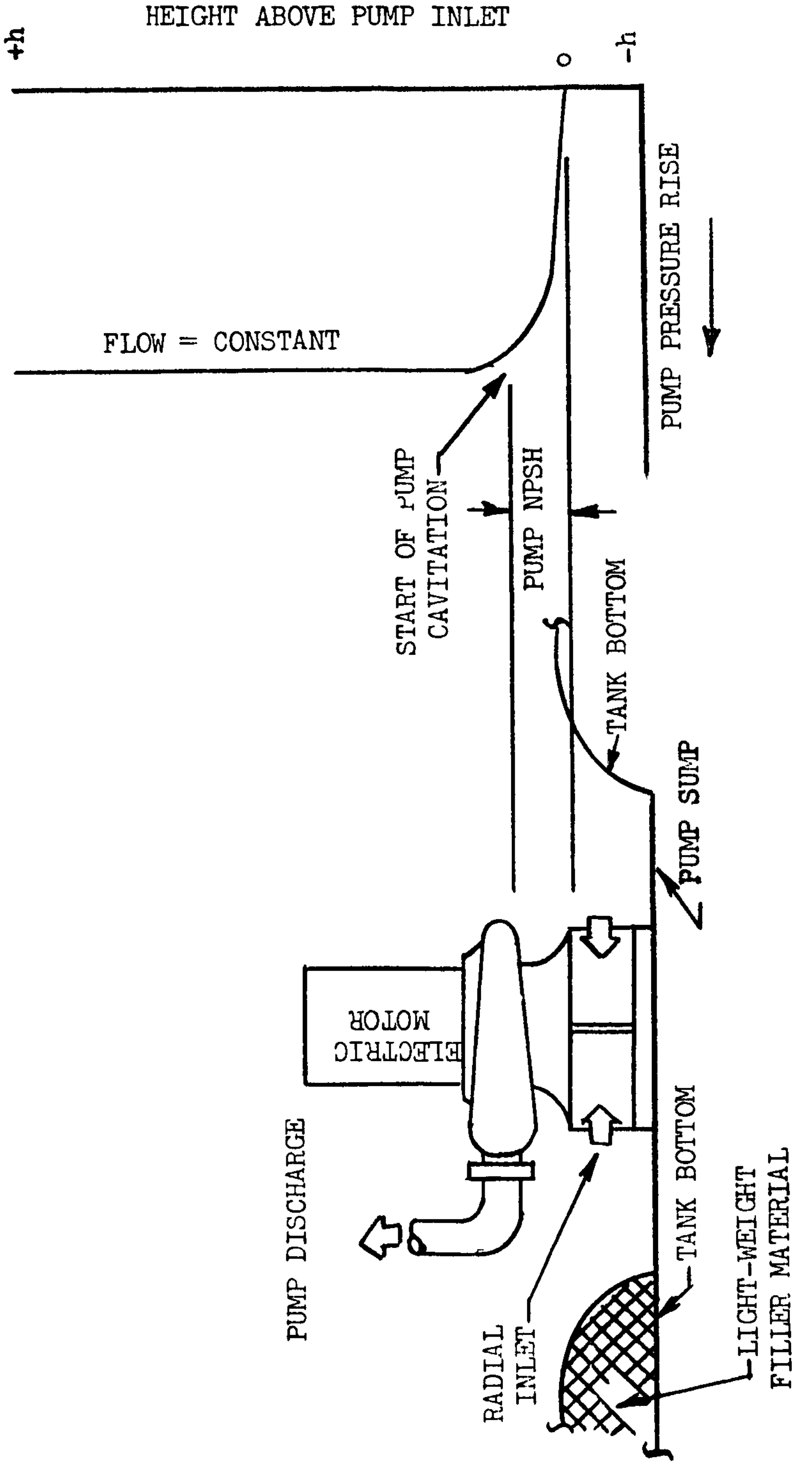

FIGURE 6.5

TYPICAL PUMP INSTALLATION TANK MOUNTED

is provided by a 2 KW (2.7 hp) alternator mounted on the RL-10 engine power take-off pad. The present RL-10 engine provides 3.6 hp at the power take-off pad.

The performance and power requirements of the transfer system are shown in Table 6-IV.

TABLE 6-IV

PERFORMANCE AND POWER REQUIREMENTS OF THE TRANSFER SYSTEM

Fluid	Performance per Pump			Total Power Requirement	
	Press. Rise psi	Flow lb/sec	Power hp	Number Pumps	Power hp
LH_2	3	2	0.6	4	2.4

Pressurization gas is admitted to the larger portion of the LH_2 tank causing the tank to become pressurized prior to engine start. The tank is pressurized in order to keep the pump pressure rise and, therefore, power requirement as low as possible. The use of electrically-driven transfer pumps becomes less attractive as the power required is increased. When the engine propellant inlet shut-off valves are opened, the pressurization system causes LH_2 to flow from the sump to the engine. Also, LO_2 will flow from the LO_2 tank to the engine. When the engine is at operating speed, the engine-driven alternator will supply the AC power to operate the LH_2 transfer pumps.

A transfer pump system is selected at this time as the best solution to a difficult feed-out problem from a toroidal shaped LH_2 tank. Since the location of the propellant surface during engine operation will attempt to be normal to the thrust vector, which can be gimbaled ± 5° in pitch and yaw, the propellant surface will tend to be parallel to the tank bottom ± 5°. Therefore, a simple pressure transfer system would be inadequate in assuring maximum feed-out of the propellants. To transfer all the propellants by pressure alone would require transfer line inlets located throughout the LH_2 tank as is done with the transfer pump system. With the transfer pump system, the pressurized manifold and pump discharge check valve prevent gas from entering the manifold. However, for a pressure transfer system, a valve and liquid/vapor sensing system would have to be provided to close off any inlet which is not covered by liquid, or else the LH_2 transfer would be replaced with gas transfer. The transfer pump system provides the best assurance of positive feed-out under the operating conditions. However, continued effort will be directed toward a gas pressure transfer system to reduce system complexity.

For proper operation of the propellant system, the propellants must be at the propellant tank outlets. Lunar gravity will position the propellants at the tank outlets for engine start prior to lunar take-off. Since the engine start for trans-Earth injection will occur while the vehicle is in an apparent zero gravitational field, a propellant settling force is provided by four 100-pound thrust reaction control engines. The 400-pound thrust is applied for approximately five seconds and provides a vehicle acceleration of approximately 2.2×10^{-2} g's.

6.4 PRESSURIZATION SYSTEM

6.4.1 Main Propellant Tank

A weight comparison was reported in Reference (3) for two methods of propellant pressurization to meet the RL-10 NPSH requirement: (1) a gas pressurization system; and (2) a tank-mounted boost pump system. The boost pump system included both electric motor and gaseous hydrogen turbine-driven boost pumps. A gas pressurization system was selected as the most desirable because the system reliability is higher, the state-of-the-art is more advanced, and total system weight, including residual gaseous propellant is less than the tank-mounted boost pump systems studied. However, a simple gas pressurization transfer system has not been worked out at this time, so a combination system is used. As previously discussed, the propellant is transferred to a sump by electrically-driven transfer pumps.

A schematic of the stored gas/hydrogen bleed (SG/HB) pressurization system is shown in Figure 6.3, which depicts the system components and pressurization gas flow path. The pressurization system is a combination system using stored helium gas and gaseous hydrogen (GH_2) bled from the RL-10 engine. Helium is stored at 4000 psia in a 25.5 I.D. sphere (4.8 ft^3 total), which is located in the LO_2 tank. Prior to engine start, helium is used to pressurize both propellant tanks to provide the RL-10 starting NPSH. During engine operation, helium is used to pressurize the LO_2 tank, while GH_2 is bled from the RL-10 engine upstream of the injector to pressurize the LH_2 tank. Redundant regulators are provided to maintain the proper helium pressure. Each regulator unit has two stages, each of which can perform the regulation function. The hydrogen bleed system uses a simple, fixed orifice to control the GH_2 bleed flow rate rather than regulate GH_2 pressure. The maximum GH_2 bleed required is 0.05 lb/sec at 270°R. An analysis has been made using preliminary RL-10 bleed pressure and temperature data, which shows that a fixed orifice system is feasible to meet the propellant system requirements. The LO_2 and LH_2 tanks are pressurized to 67 psia and 48 psia respectively to provide RL-10 engine starting and operating NPSH and to overcome the estimated LO_2 and LH_2 line losses of 7 psi and 3 psi respectively. The helium pressurant required is 29.7 pounds. The selected storage pressure of 4000 psi is based on a compromise between system weight and volume. System weight can be reduced slightly by reducing storage pressure, but system volume is increased rapidly. The pressurization system weight is based on helium storage spheres made of 301 stainless steel, 60% cold rolled, ultimate stress at 174°R of 200,000 psi. The strength-to-weight ratio is 0.7 psi/lb/in^3. The use of titanium spheres as shown in Reference (3) was eliminated because titanium is not compatible with LO_2. This material change increased the system weight approzimately thirty pounds. A search of materials which can be used in LO_2 for high pressure spheres should reveal several other materials which have a strength-to-weight ratio greater than 0.7. The above system and pressurant weights are based on data presented in Reference (11). Additional system detail can be obtained from Reference (3).

6.4.2 Engine Pneumatic System

The RL-10 engine requires helium at 470 $\pm$ 30 psia and 300°R to 600°R to operate engine pneumatic valves. For the lunar take-off stage, 0.21 pounds of helium is required to provide for two engine starts and shutdowns and valve leakage, Reference (7). Helium is isolated from the engine by squib balves until engine start for lunar take-off. By isolating the helium from the engine, helium leakage is greatly reduced. A schematic of the system is shown in Figure 6.3. Redundant regulators and squib isolation valves are provided to assure that helium is available at the engine. Each regulator unit has two stages, each of which can perform the regulation function.

6.5 REACTION CONTROL SYSTEM

A reaction control system (RCS) is required on the direct manned lunar landing vehicle to provide translation and rotation maneuvers during the translunar, lunar landing, lunar take-off, and trans-Earth mission phases. The anticipated maneuver functions which must be performed by the RCS are:

1. Arrest tumbling resulting from uneven shut-down of the main landing engines, or possible misalignment of the thrust vector through the vehicle c.g. on any engine shut-down.

2. Navigational sighting orientation maneuver in which the vehicle is rotated about its roll axis to the desired orientation. The angular maneuver rate is then reduced to a dead band velocity to allow for navigational sightings.

3. Δ V orientation maneuver used to orient the vehicle about any axis to the desired orientation prior to and after ΔV corrections are made by the main engine.

4. Miscellaneous orientation for maneuvers such as orienting for coast after translunar injection.

5. Roll control during lunar landing, allowing a vernier roll or precise orientation of vehicle during hover, let-down, and flare of the lunar landing operation. Roll control is required of the RCS during lunar take-off due to the single main engine configuration.

6. Propellant settling thrust to ensure main engine starts.

7. All mid-course corrections required during the trans-Earth mission phase will be made using the RCS. It is also possible that small ΔV corrections will be made with the RCS to cope with any velocity errors that may be induced by manual control of the main engine.

8. Command module separation from the take-off stage will require the RCS to turn the take-off stage 90° after separation.

9. Attitude hold of the vehicle to keep the vehicle oriented in a manner to ensure vehicle-Earth and vehicle-Moon communication at all times.

6.5.1 System Installation

The RCS, including thrust chambers, will be located on the take-off stage. This configuration will allow one basic system to be used for all phases of the mission before and after staging. The RCS, divided into four independent systems, will have a total of sixteen thrust chambers; eight roll, four pitch, and four yaw. Four engines will be mounted in a cluster on the external surface of the vehicle. Control in each axis will be supplied by pairs of engines arranged to provide rotation in that axis with minimum translation. The chambers are canted out slightly to minimize engine exhaust impingement on the vehicle surface adjacent to the cryogenic propellant tanks. During the translunar phase, in which the vehicle configuration includes the lunar landing stage, the use of the roll thrust chambers for pitch, yaw, and roll control may be possible because of the aft location of the c.g. Using the roll chambers for control about all three axes minimizes the problem of jet impingement. However, propellant requirement would be increased using the above method due to shorter moment arms. During lunar launch, or after staging, the exit planes of the aft firing RCS nozzles are located at the approximate stage separation plane, thus eliminating the exhaust impingement from these particular engines. At lunar take-off, the vehicle c.g. location is in the approximate station plane of the thrust chamber location, allowing a true couple about each axis.

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6.5.1.1 Engines

The RCS will consist of pulse-modulated, radiation cooled, pressure-fed rocket engines in the 100-pound thrust class. The engines will use Earth storable bi-propellants of the type used in the present Apollo and Gemini. Propellant control valves located on the engine will provide pulse width control. Separate redundant coil windings for each solenoid valve will allow manual control by the crew.

6.5.1.2 Propellant Feed System

The complete RCS consists of four identical systems, one located at each quadrant of the take-off stage. (Refer to Figure 6.6.) The four identical systems are capable of operating simultaneously. During normal operation, only two systems would operate simultaneously for each axis of control. A schematic of one of the four systems is presented in Figure 6.7. Each propellant system consists of a pressurized helium storage and distribution subsystem, and a fuel and oxidizer storage and distribution subsystem. The high-pressure helium is regulated by two stage regulators located in parallel. One stage of any regulator is capable of reducing the pressure to the required working level of the propellant feed system. Positive expulsion bladders or metal diaphragms are used in the propellant tanks to insure engine operation during zero g conditions. The system is designed to operate automatically by means of electrical inputs which sequence the system operation.

6.5.1.3 System Configuration

Three basic system configurations were studied to determine an optimum system considering weight, reliability, and system feasibility. Reference (2) discusses these configurations in detail. Some advantages gained

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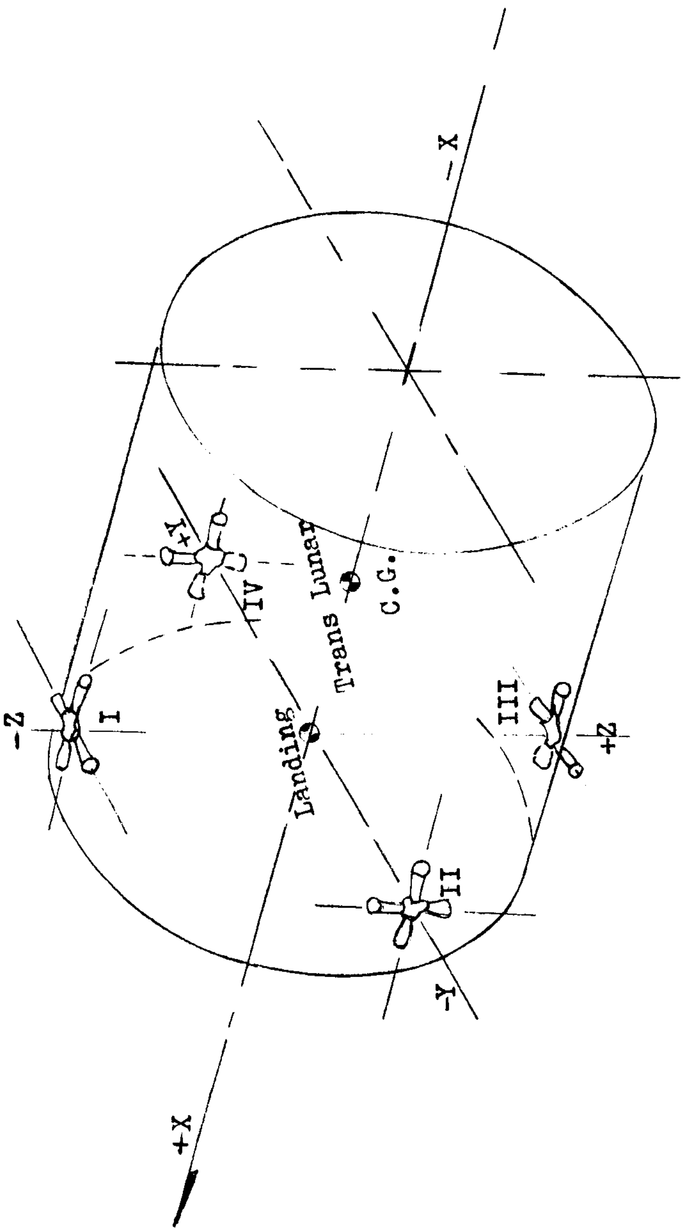
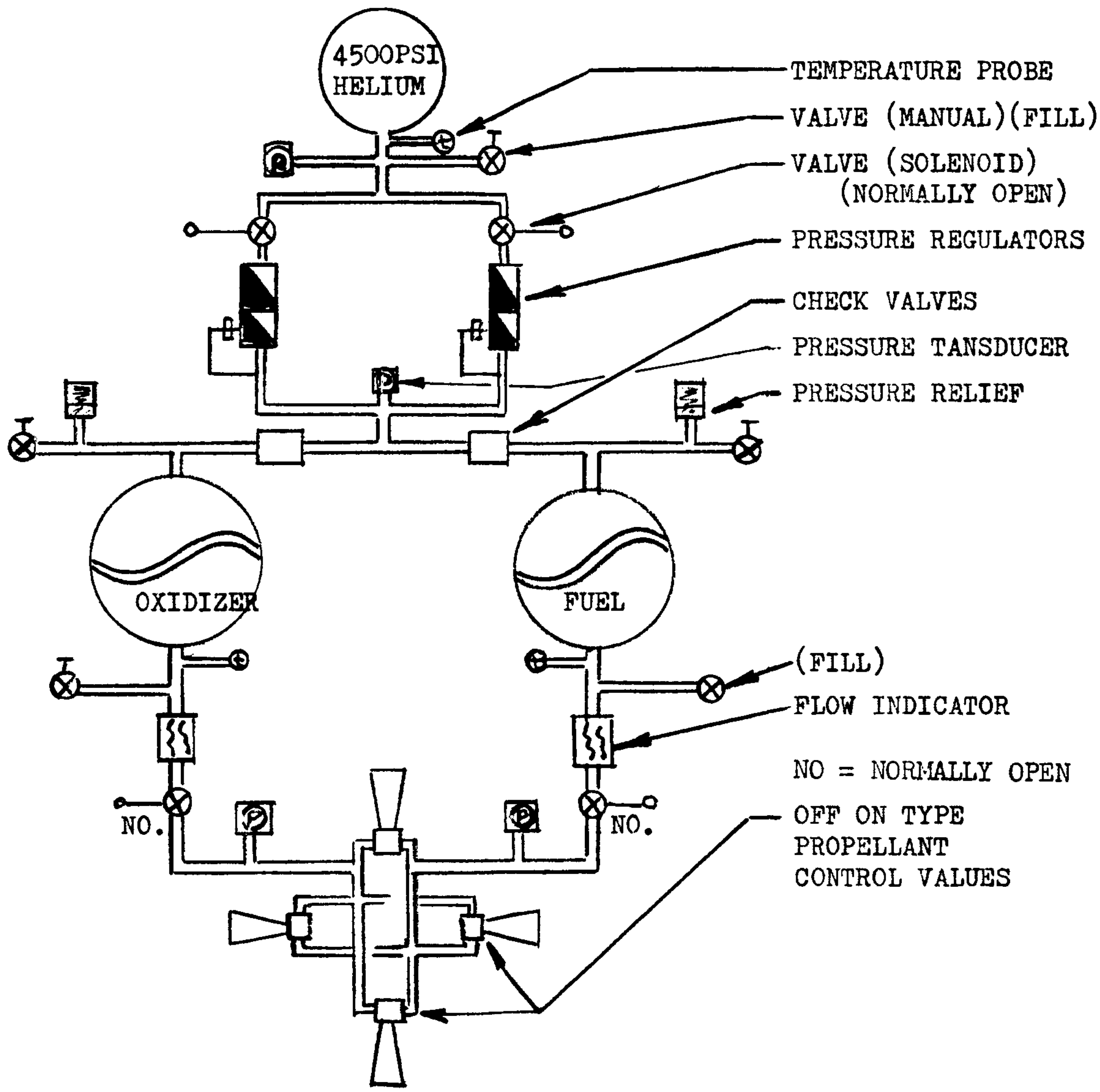


Figure 6.6 REACTION CONTROL SYSTEM - THRUST CHAMBER LOCATION



ONE OF FOUR SYSTEMS

Figure 6.7 REACTION CONTROL SYSTEM SCHEMATIC DIAGRAM by using the four-system configuration are simplicity in feed system design, fabrication and checkout maintained by the manufacturer prior to delivery, and easy launch pad logistics.

6.5.1.4 Propellant Requirements

To enable realistic propellant requirements to be made, the total number of maneuvers during each phase of the mission has been estimated. Using the various maneuvering functions described, the possible number of maneuvers, the slewing rates, V's required, etc., were estimated, allowing the total impulse to be determined. Confidence in these estimates is based on experience with the present Apollo RCS duty cycle. A breakdown of the RCS propellant required per function for each mission phase is presented in Table 6-V.

To insure high mission reliability versus minimum weight, redundant propellant is included for failure mode operation. The following assumptions were made to determine the redundant propellant required.

If a failure occurs in RCS after translunar injection, lunar orbit injection, and Hohmann transfer, but prior to lunar retro descent, no redundant propellant is required since the landing module is jettisoned and Earth return is made, using the RCS propellant which is normally available for lunar landing and take-off. Therefore, the propellant for the translunar operation is divided equally among the four systems. Translunar mission phase presented in Table 6-V represents all functions from translunar injection to retro descent; lunar landing covers retro descent to lunar touchdown, and trans-Earth phase includes all functions from lunar lift-off through command module separation prior to Earth re-entry.

After the lunar landing operation has started, sufficient redund-

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TABLE 6-V

ESTIMATED PROPELLANT WEIGHT - LB

FUNCTION \ PHASE	TRANS-LUNAR	LUNAR LANDING	TRANS-EARTH	TOTAL PER FUNCTION	TOTAL PER SYSTEM	MISSION TOTAL
Arrest Tumbling	2.0	0	0.6	2.6		
Navigational Sightings	54.5	0	5.7	60.2		
Δ V Orientations	32.6	0	3.1	35.7		
Misc. Orientations	3.4	0	0.8	4.2		
Roll Control (landing & takeoff)	0	28.5	6.6	35.1		
Propellant Settling	36.3	0	3.5	39.8		
Mid-course Correction	0	0	442.0	442.0		
CM Separation	0	0	5.0	5.0		
Attitude Hold	1.0	50.0	5.0	56.0		
Single System Total	129.8	78.5	472.3		680.6	680.6
Two Independent Systems	64.9	78.5	472.3		615.7	1231.4
Four Independent Systems	32.5	30.8	234.5		297.8	1191.2

NOTE: Propellant required for function only - trapped propellant not included.

*Defined as start of retro descent

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ant propellant is included to accomplish a safe landing and return with one of the four clusters inoperative. Therefore, any three clusters must provide the propellant for all maneuvers except translation, for which any two clusters must be able to perform the necessary functions.

For an example, refer to Figure 6.6. Assume cluster IV is inoperative, cluster II will only be used to provide yaw attitude control. Therefore, cluster I and III must provide all other attitude control and translation functions. Propellant requirements were based on this type of failure mode operation.

6.5.1.5 Propellant Selection

Total impulse requirements plus present state-of-the-art in designs of reaction control system favor the use of a bi-propellant system. Because of attendant system complexity, and numerous on-off operations of the control system, the use of the non-hypergolic, main engine propellant, LO_2/LH_2, is precluded. For this reason, hypergolic Earth storables were chosen. Nitrogen tetroxide and a 50-50 mixture of UDMH and hydrazine are presently being considered; however, a mixed oxide of nitrogen/monomethylhydrazine combination appears to be an alternate choice because of its desirable operating temperature range. This propellant combination is interchangeable with N_2O_4/50-50 UDMH-N_2H_4 without modification of the basic propulsion system design. Physical properties of these propellants are shown in Table 6-VI.

6.6 LANDING STAGE SUMMARY

As mentioned in the introduction to the propulsion section, the landing stage propulsion of the direct manned lunar landing vehicle is identical to the landing propulsion for the lunar logistics vehicle (LLV). This

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TABLE 6-VI

REACTION CONTROL SYSTEM

PROPELLANT PHYSICAL PROPERTIES

PROPELLANTS	BOILING POINT °F	FREEZING POINT °F	DENSITY AT 68°, g/ml	VAPOR PRESSURE 68°F PSIA	VAPOR PRESSURE 160°F PSIA
N_2O_4	70	12	1.44	14	111
MON (75% N_2O_4 - 25% NO)	18	-61	1.381	52	370
50-50 UDMH - N_2H_4	152	19	0.903	2.00	15.3
MMH	190	-62	0.877	0.72	7.9

[redacted]

section provides a summary of the landing stage propulsion system. For additional details, refer to the LLV report of Reference (1).

The main propulsion system provides thrust for accomplishing the trans-lunar mid-course correction, lunar orbit injection, Hohmann transfer, retro descent, and landing phases of the mission. Main propulsion is provided by two 15,000 pound thrust, pump-fed engines which can be throttled to 10% of rated thrust for a controlled landing on the lunar surface. The engines are gimbal-actuated to provide for thrust vector control. In the normal null position, the engine thrust vectors are parallel. Electro-mechanical gimbal actuators provide for pitch and yaw control to $\pm$ 5°. During the terminal landing maneuvers, the engines are canted 15° in the yaw plane. In this stepped null position, the thrust vectors pass through the vehicle c.g., and the elctro-mechanical actuators control pitch and yaw to $\pm$ 5° about this new position. The canted position of the engines provides improved control by eliminating any tumbling of the vehicle if an engine should fail during the critical landing phase.

The liquid oxygen and liquid hydrogen propellants are fed to the engines from two toroidal-shaped propellant tanks. The tanks are compartmented to form a sump area, from which the propellants flow directly to the engine. Propellant transfer to the sumps is provided by electrically-driven transfer pumps. The sump compartments are sized to provide sufficient volume for all of the propellant required for the terminal landing maneuvers. These compartments serve to localize the propellant, and, together with the transfer pump system, insure that the last remaining quantity will be fed from the large toroidal tanks when the vehicle is experiencing attitude changes during the hover-translation-letdown phases of landing.

The pressurization system is similar in concept to that used in the take-off stage. Pressurization of the propellant tanks is provided by a supply of helium, stored at 4000 psi in a pressure vessel that is submerged in the liquid oxygen. During engine operation, the fuel tank is pressurized by gaseous hydrogen bled from the engine. Prior to starting the main engine, propellant settling is provided by operation of the reaction control system in the translational mode.

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7.0 ELECTRICAL POWER SYSTEMS

The power system for the direct-mission Integrated Apollo capsule has been analyzed with the objective of obtaining minimum weight without sacrificing reliability. The analysis has been based on the evaluation and design studies presently going on for the LEM version of the Apollo.

The reassessment of the load requirements of the integrated version indicates that power and weight reductions are possible. The power has been reduced by careful re-evaluation of the power profile and the needs of the single units. Weight reduction has been accomplished mainly by using microminiature electronic components. The power requirements of the integrated Apollo capsule vary now between I.0 KW and 2.0 KW with peaks of very short time duration.

The primary power source of the LEM version has also been reanalyzed with the new conditions in mind. The evaluation has been extended to all applicable systems because of the new conditions and the possibility that improvements may have occurred since the first evaluation of the LEM system.

As a result of the integrated Apollo studies, the choice of a power source has been narrowed to (1) Fuel cell systems, or (2) Isotope dynamic systems. A detailed evaluation of these systems is presented in this report.

7.1 POWER SOURCE ANALYSIS

Power Sources can be divided into three basic kinds:

a. Sources consuming chemical fuel

b. Sources using solar energy

c. Sources using nuclear or isotopic energy

These systems can be combined or integrated with other systems of the spacecraft; for instance, the environmental or the propulsion system.

The sources which consume chemical fuel exhibit many variations. They can differ in the kind of cycle they operate, as Rankine cycle, Ericcson cycle, and Brayton cycle; or in the types of engines, as turbines or reciprocating engines, and in the type of combustion. Many systems of various vendors have been studied, but they were discarded because of two reasons; the hardware was in an early development state in most cases, and the fuel consumption is high compared to the fuel cell.

The fuel cell power systems presently designed for the Gemini and for the Apollo are in an advanced development state or nearly state-of-the art and should be considered for selection. Both systems use hydrogen and oxygen as fuel; both produce drinkable water.

The membrane type fuel cell developed for the Gemini operates at relatively low temperatures and needs relatively large radiator surfaces for operation on the lunar surface. The radiators have to be of special design to dissipate heat on the Moon, and they have to be deployed for best efficiency after landing. Several reasons why the membrane-type Gemini fuel cell was not selected are:

1. Efficiency of the Bacon-type fuel cell is higher at comparable specific load ratings. That means for the same efficiency in operation, a smaller fuel cell plate area is needed.

2. The membrane-type fuel cells have a poorer voltage regulation than the Bacon-type fuel cell. It needs an additional series voltage regulator which reduces the system reliability. Elimination of the voltage regu-

lator was considered but it would have resulted in a considerable increase in complexity of equipment operating from the fuel cells.

3. The Bacon-type cell operates at higher temperature and, therefore, needs less radiator surface and can operate without complication on the Moon surface.

4. The operating temperature of the Bacon-type fuel cell can be varied with the load with a resulting improvement in voltage regulation and efficiency at higher load ratings. The voltage regulation of the Bacon-type fuel cell can be held with 10% of the set value for steady state conditions.

5. Improvements in fuel cell technology reduced the Bacon-type fuel cell weight and increased its efficiency.

Based on these considerations, the Bacon-type fuel cell system has to be favored. In the following proposal, two fuel cell power systems will be presented using Bacon-type cells. One is a simplified dual redundant fuel cell system using less space and power plant weight than two individual fuel cell power plants. The other fuel cell power system uses triple standby redundancy in cells and uses one fuel cell power plant at a time. The latter is highly reliable with comparable reliability values with the three parallel operating fuel systems. The second system is less efficient than the first, but it weighs less and has higher reliability; this is our recommended design.

Besides the fuel cell systems discussed, other systems are also on the market, but not in an advanced stage of development. The most notable is the dual membrane fuel cell system which has much promise but needs considerable development.

A very special case is the type of power source of which the

Sundstrand cryhocycle is an example. These systems use the heat dissipated in the capsule. They operate on an open cycle and use hydrogen as a heat sink. This system has been analyzed very thoroughly by recalculating the cycle conditions as explained by the vendor and specifically evaluating two important parameters - the polytropic coefficient and the mean effective friction coefficient. Our analysis showed that, with conservative assumptions, the fuel consumption would be nearly twice as high as for the fuel cells and as the values given by the vendor. The boil-off of the hydrogen tank is too low to be taken into consideration as a source for this system. In addition to that, other factors influenced the decision to discard the system. The hardware is not off the shelf and still is in an early development state. Integration of different systems does not seem advisable due to the uncertainty of the environmental conditions during the different phases of the mission.

The weight of solar power systems is comparable to the weight of the fuel cell systems. Solar cells, however, vary in the output voltage and current with the incident radiation and ambient temperature. On the Moon surface, the average temperature of the cells is not the result of the balance between the incident radiation from the Sun and the radiation to outer space as is true for vehicles in interplanetary orbits. The reflected sunlight from the Moon surface raises the cell temperature considerably and greatly reduces the output. Solar cells need additional mechanisms for continuous orientation to the Sun and deployment. They also need provisions for voltage regulation, energy storage and battery charging, not to mention problems of shock conditions encountered during landing on the Moon and during lunar take-off. Any and all of these problems can be solved and may result in a reasonably reliable system. Yet, for the early missions, the solar cell concept has been discarded because of the uncertainty of the landing site on the Moon. There is no guarantee that, for one reason or another, the vehicle may not land in an area where no sunlight is available at all. That may occur if the spacecraft lands on the dark side of the Moon, in the shadow of a mountain, or in a rill. Yet, the system should be reconsidered for later missions or logistic systems.

Solar thermoelectric and solar thermionic systems exhibit the same major disadvantages as just mentioned for the solar cells. They are in a considerably earlier state of development than the solar cells and much additional work has to be done until they are actually feasible for such a delicate mission.

Nuclear fission power sources are much too heavy compared to other systems. The reason for this is the necessary radiation shielding. The shielding weight can be considerably decreased for nuclear-isotopic power sources; this type of source has been seriously considered as an alternate and probably advantageious solution, compared to a fuel cell system. Besides the weight saving, this system has the distinct advantage that the mission time could be extended considerably. Two questions have to be considered if such a power system is considered; the availability of the isotope and the secondary power system. The power generated is AC instead of the DC power from the fuel cells or other power sources, and this has to be compatible with the load and the storage system.

An isotopic dynamic power system using the Brayton cycle has been chosen as the second recommended power source.

7.2 FUEL CELL POWER SYSTEMS

7.2.1 Recommended Fuel Cell Power Design

The fuel cell power plant will be designed for partial stand-by redundancy. (In contrast with the present design, where several fuel cell systems operate in parallel at the same time, only one fuel cell system operates.) For redundancy and high reliability, all essential subassemblies are duplicated. As explained later, some of the duplicated sub-assemblies are continuously operating; some of them are on stand-by. This arrangement will have comparable reliability with the present Apollo fuel cell system and will have less weight and requires less space. See Figure 7.1.

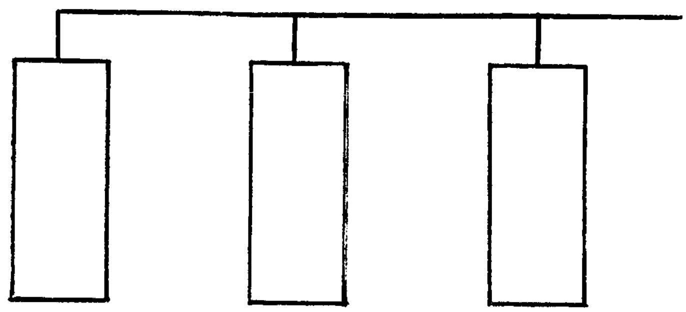
The basic fuel cell system is based on the present Apollo fuel cell design using the latest version of cells. This arrangement will use thirty cells in series connection and will maintain 28V $\pm$ 2V under a load range of 1 KW to 1.8 KW. See Figure 7.2.

The minimum load on the system under normal conditions is 1.0 KW with 29.3V output. Under maximum load condition of 1.8 KW, the voltage will drop to 26.0 volts. If the load would increase beyond 1.8 KW, the storage batteries would be connected automatically and would share the load with the fuel cells until the load current decreases to a value corresponding to 1.5 KW load. This arrangement will maintain relatively constant voltage on the D.C. line and will operate the fuel cells at high efficiency.

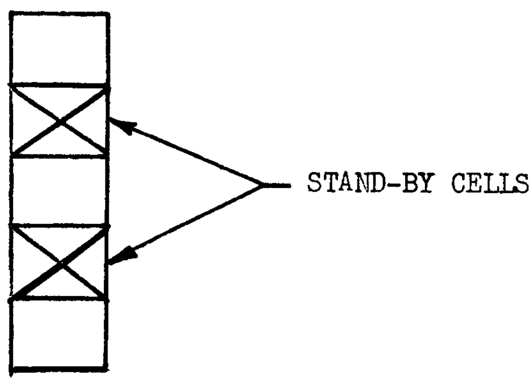
7.2.1.1 Arrangement of the Cell System

The fuel cells are arranged into five sections, with ten cells in each section. The five sections are in one assembly but have individual

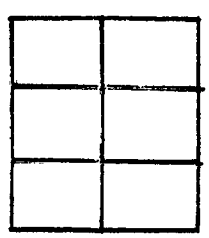
CON[redacted]



PRESENT APOLLO FUEL CELL ARRANGEMENT
3 FUEL CELL SYSTEMS ARE OPERATING
ALL THE TIME IN PARALLEL CONNECTION



PROPOSED FUEL CELL ARRANGEMENT WITH
STAND-BY CELLS INSERTED BETWEEN
OPERATING CELL TO PROVIDE EASY START-UP



ALTERNATE ARRANGEMENT WITH TWO FUEL CELLS IN
ONE PACKAGE EACH SECTION CAN BE DIS-CONNECTED
SEPARATELY COMMON AUXILIARY EQUIPMENT IS USED

FIGURE 7.1

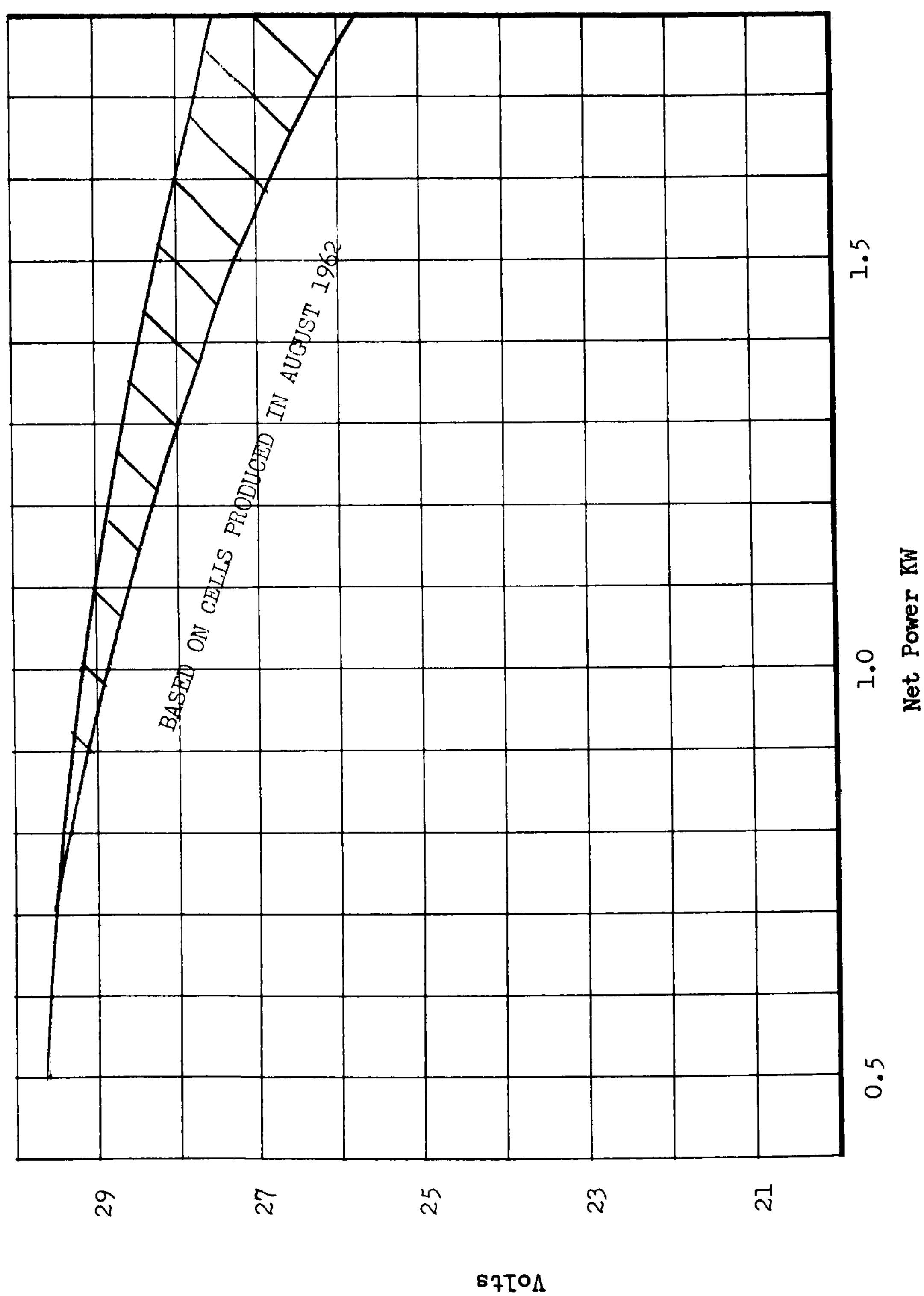


FIGURE 7.2

PERFORMANCE OF A 1.35 KW POWERPLANT USING 30 P&W FUEL CELLS IN SERIES connections to both the electrical system and to the fuel system. Under normal operation three sections are connected in series while two additional sections are on stand-by. Only three groups generate voltage. Under any failure of one section, the stand-by cells are connected into the system and the faulty section is isolated. The active cells provide enough heat for the stand-by cells to maintain stand-by redundancy instead of the present parallel redundancy, increasing the system over-all reliability.

The hydrogen regenerator, reactant regulators and coolant system are designed to be redundant.

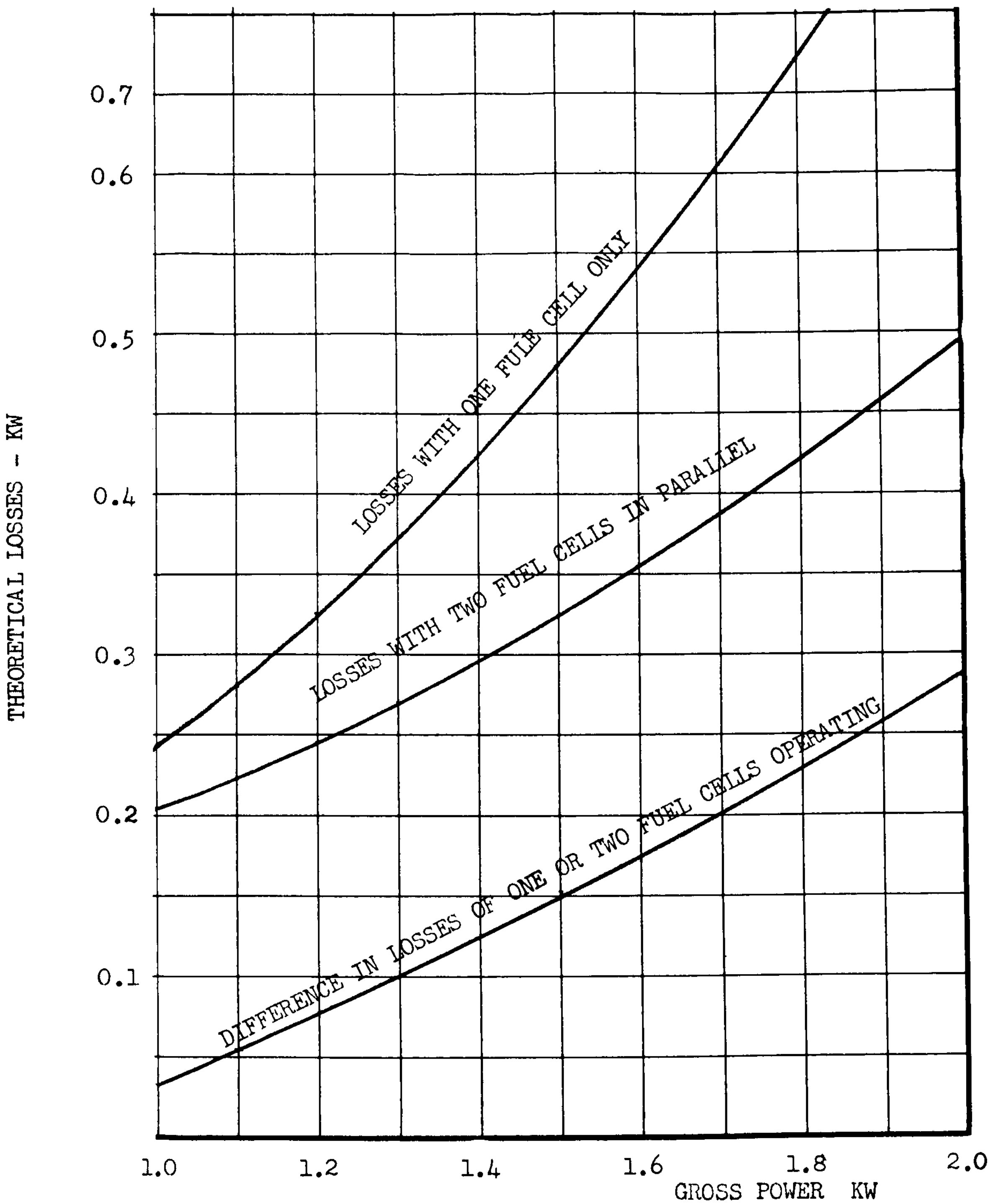
There will be a complete stand-by hydrogen regenerator assembly which is switched on as soon as the power to the separator pump motor is disconnected.

The reactant regulators are connected in series and the main regulator operates in tandem with the stand-by regulators. If one regulator fails, the other still will maintain the pressure within range of specification.

There is a complete stand-by coolant circuit which under normal conditions is not in use. Only when fuel cell temperature exceeds a certain value would this circuit be activated.

The fuel cells considered are identical to the cells presently in production and the design is based on the test data obtained from Pratt-Whitney. It is, however, expected that the fuel cell efficiency actually will be improved over the present data.

In the attached diagram, Figure 7.3, the losses of the fuel cells are plotted. These are the losses of the electro chemical conversion only, and do not include the losses of necessary coolant pumps and water separation.



NOTE: Data presented here are pessimistic and based on poorest fuel cell tested to date.

<u>FUEL CELL LOSSES</u>

FIGURE 7.3

Considering these losses to be 100 watts for each fuel cell system, the break even point for best efficiency can be calculated. Presently the break even point between the use of one or two parallel operating fuel cell power plants is 1300 watts. If less power is consumed, one cell alone is more efficient. For more power, 2 parallel operating fuel cells should be selected. On the graph, the difference of losses is plotted using one and two fuel cell systems. With advances in fuel cell technology, the use of one fuel cell power plant only instead of two in parallel is more and more advantageous and considering an average load of 1200 watts during the mission some weight saving is obtained. See Table 7-I.

The use of stand-by redundant sections requires start-up of cells during mission. Stand-by cells are filled with inert gas (nitrogen or equivalent). To activate these cells the inert gases have to be flushed out. During flushing of the cells, some extra fuel is consumed. It also takes a short time to start up the cells. During start-up of the stand-by cells, two methods can be used:

(1) Switching to the storage batteries which can be activated when the voltage drops below 25V;

(2) Reducing the power requirements to emergency level and operate using the faulty cells.

The chances are practically nil that a fuel cell would fail open circuited. If a cell is short circuited, the power supply voltage drops approximately by 1 volt, but it is still operative. A simple check of voltage output of each section would reveal the location of the fault.

7.2.1.2 Weight of the Fuel Cell Power System

The weight of the fuel cell power system used for the Apollo was

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TABLE 7-I

POWER REQUIREMENTS FOR A TYPICAL MISSION

NOTE: Values are given in watts (peak values in bracket)

		Ascent	Earth Orbit	Trans-Lunar Inject	Trans-Lunar	Moon Orbit	Moon	Moon Ascent	Trans-Earth Injection	Trans-Earth Coast
		9 Min.	21 Hrs.	6 Min.	70 Hrs.	12 Hrs.	168 Hrs.	6 Min.	2 Min.	65 Hrs.
Illumination		150	150	150	100	100/150	20	150	150	100
Propulsion	150			(150)	(150)			(150)	(150)	(150)
In-Flight Test			(100)		(100)	(100)	(100)			(100)
Guidance & Navigation		210	274	210	50	210	50	210	274	50
Stabilization & Control		202	202	202	100	100	50	202	202	100
Telecommunication		182 (202)	182 (202)	112 (192)		112 (192)	40 (242)	112 (192)	112 (192)	112 (192)
Displays		100	100	100	100	100	70	100	100	100
Environmental Control		180	209	180	180	209	209	180	180	180
Auxiliary Power		210	210	210	210	210	190	190	210	210
Total		1234	1327	1164	740	1041	629	1144	1238	852
Scientific Instruments & Pad for Design Variations		200	200	200	400	200	400	200	200	200
Total - including 6% Dist. Losses		1520	1620	1450	1210	1320	1090	1420	1530	1120
Total KW Hours			34		85	15.8	183			73

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originally estimated to be 245 pounds. Three redundant power systems are used in parallel. Since the original estimate improvements in technology and weight reduction of components brought the weight down to 215 pounds. An additional weight saving is expected by reducing the size of the fuel cells for the parallel operating systems. The smaller cells would still have the same efficiency as the worst cells now in test. This reduction would bring the weight of the individual power system to 185 pounds as presently used for the Apollo.

Weight of the proposed power system would be more than the weight of one of the presently used power systems but less than two and considerably less than the presently used power system package of three parallel redundant fuel cells.

Estimated weight of the new system using 30 cells in series and 20 stand-by cells all using the present electrode area is calculated in Table 7-II.

7.2.2 Alternate Fuel Cell Power System Design

There is another arrangement of fuel cell systems which may result in weight saving. This arrangement packages two complete groups of 30 fuel cells into one module jacket and operates the two fuel cell systems in parallel all the time. The controls are redundant, mostly stand-by redundant. The arrangement of controls is similar to the system previously presented.

The fuel cell is subdivided into sections of 10 cells and two groups of 10 cells operate in parallel. In the case of failure of one cell in a section, the parallel section can handle the entire load. The faulty section can be severed both electrically and pneumatically.

TABLE 7-II

FUEL CELL POWERPLANT SYSTEM ESTIMATED WEIGHT BREAKDOWN

	Unit Weight - Lbs.
Fuel Cell Assembly	
Cells (50)	121
Heaters	8.3
Bolts and Tension Bars	10
Tubing and Fittings	5.5
KOH	38
	182.8
Module Jacket	
Inner Jacket	17
Mount Ring and Outer Jacket	4.8
Insulation	9.1
Plumbing	2.4
Miscellaneous	4.3
	37.6
Hydrogen Circuit	
Regenerator - Primary Loop (2)	31
Condensor (2)	10.5
Separator Pump Motor Assembly (2)	22.5
	64.0
Controls	
Reactant Regulators (2 x 2)	3.0
Inert Gas Regulator (2)	1.6
Regenerator Bypass	3.8
Vent Valves (6)	3.9
	12.3
Glycol Secondary Circuit	
Pump Motor (2)	5.3
Regenerator (2)	31.0
Accumulator (2)	5.5
Regenerator Bypass	1.7
Fill and Relief Valves	1.0
Plumbing	1.7
Support	2.5
	48.7
Other	
Reactant Preheaters	1.0
Inert Gas Tank (2)	1.0
Tubes and Fittings	8.0
Mounting	4.5
Electric Connectors	5.0
	19.5

TABLE 7-II (Contd)

TOTAL MODULE WEIGHT	364.9 lb

Weight Estimate for the Electrical Power System in the Service Module:

Fuel Cells	365
Liquid H_2	41.4
Liquid O_2	329
Tankage	30
Radiator	40
Hardware	20
Fuel Pumps	15
Total	840.4 lb
Wiring & Power Distribution	78 lb
Pyrotechnic Initiations	45 lb
TOTAL POWER SYSTEM WEIGHT IN THE S/M IS:	963 lb

Estimated Weight of Electrical Power System in Command Module:

Batteries	70
Chargers	4
Inverters	30
Distribution and Control	72
Wiring - Power Distribution	25
Electric Hardware	12
Pyrotechnic Initiation	43
Relays	68
GSE Wiring	22
Total	346 lb
TOTAL WEIGHT OF THE POWER SYSTEMS IN BOTH THE S/M AND C/M IS:	1309 lb

*This weight compares favorably to the original Apollo using three fuel cells, each weighing 215 pounds.

This arrangement takes less weight than two completely independent fuel cell systems and also takes less room. It has the same reliability as two completely independent fuel cell systems. The system also operates at higher efficiency than the recommended fuel cell system design.

This system, however, is inferior to the recommended system in the following respects:

1. It has more voltage regulation, considering all the different modes of operation.
2. It has lower reliability; operating redundancy is used instead of stand-by redundancy.

The advantages of the system are:

1. Better efficiency
2. No start-up problems of stand-by sections where failure occurs.

Since this arrangement uses slightly smaller diameter fuel cell plates, the kind of plates which are intended for final use in the present Apollo design, the weight of this system and that of the previously presented are comparable.

7.3 ISOTOPE DYNAMIC POWER SYSTEM

Nuclear power systems which make use of alpha emitting isotopes as Plutonium, Polonium, or Curium need much less shielding so that the total system weight including redundancy may be reduced to approximately 300 to 600 pounds. Based on an efficiency of about 17% to 30%, a thermal power of 5-8.5 KW will be needed. The amount of fuel material needed can be provided by A.E.C. There is, however, some effort required for increased production. A preliminary study of the isotope boiler is being performed both by Atomics International and NAA/S&ID. A study of the thermodynamic system including turbines, heat exchangers and radiators is being performed by NAA/S&ID in connection with AiResearch. Preliminary specifications of this system have been worked out and integration problems with the Apollo system considering prelaunch period, aerodynamic heating and radiation problems during stay time on the moon are under investigation. The design uses only proven parts except for the isotope boiler, which is similar to existing boilers and would need relatively little development.

A separate report regarding this isotope dynamic power system is being prepared.

Our feasibility study showed that only two primary power systems can be used: (1) Fuel cell and (2) Isotope turbine system. Our conclusion is that isotope turbine systems seem to have a great advantage and shall be favored. The fuel cell is an alternate choice, since it is developed for the Apollo and would not introduce new problem areas.

The power supply as discussed in the report by W. K. Luckow is rated 1300 watts with a weight of 580 pounds. The power supply can furnish more power if radiator temperatures are below the design temperatures calculated for lunar mid-day conditions. On the moon, the power requirements are only about 1,000 watts. In space, the radiators are more effective and the system can handle considerable overload.

It can be safely assumed that batteries are only needed for sudden peak loads or for re-entry.

Most of the power supplied has to be rectified since the command module electrical system has to operate from batteries after re-entry into earth atmosphere.

The power system voltage can be selected such that the rectified DC voltage is in the range of 28V + 2V.

Weight estimates for the electrical power system in the service module:

Isotope Dynamic Power Supply	480 lbs.
Wiring and Power Distribution	78 lbs.
Pyrotechnic Initiations	30 lbs.
	688 lbs.

Weight estimates for the electrical power system in the command module:

Batteries	70
Chargers	4
Rectifiers	5
Distribution and Control	72
Wiring - Power Distribution	25
Electric Hardware	12
Pyrotechnic Initiation	43
Relays	68
GSE Wiring	22
Total	320 lbs.

The total weight of the power systems in both the service and the command modules is 1008 pounds.

This system needs 250 pounds of drinking water and an additional ten pounds of water for the ECS system operation on lunar mid-day. The power system and water weight is 1268 pounds.

The isotope dynamic power system as designed has no redundant components. For higher reliability, some of the power system components should be duplicated. This may increase the system weight.

7.4 SOLAR PHOTOVOLTAIC POWER SYSTEM

The power system considered was discussed by D. C. Stager in his recent report dated 25 September 1962. The power rating of this system was reduced to the required 1300 watt level to make it comparable with the fuel cell system and the isotope dynamic system. The revised photovoltaic system has a power rating of 1090 watts under lunar mid-day conditions, considering the reflected heat of the moon surface. Under other load conditions, the system can furnish somewhat more power than needed for average load.

The weight estimate has to be based on the type of storage batteries used in connection with the solar cells. The worst condition is earth orbit, for the batteries need more energy storage there than at any other time.

Weight estimate of electrical power system in service module:

Battery Weight	AG-Zu	Ni-Cad	Ag-Cad
	222	93	96
20% Battery Redundancy	45	19	20
Charge Regulator	20	20	20
Series Regulator	30	30	30
Solar Panels	326	294	329
Power Distribution	78	78	78
Pyrotechnic Initiation	45	45	45
Total	766	579	618

Weight estimate for command module:

Inverters	30 lbs.
Distribution and Control System	72
Wiring	25
Electrical Hardware	12
Pyrotechnic Initiation	45
Recovery Battery	33
Relays-Power Distributions	68
GSE Wiring	22
Total	207 lbs.
Drinking Water	250 lbs.
Water for ECS System	10 lbs.
	260 lbs.

Total System weight using Ni-Cad Batteries 1146 pounds.

The solar panels discussed are designed to withstand vibration loads encountered during flight, engine firing, etc., but are not designed to stand shock or impact loads when deployed. For lunar landing, they have to be folded and specially enforced. It is expected that this special equipment may add to the weight of the solar panels. The cost of the solar panels is also extremely high compared to other power system solutions.

7.5 WEIGHT COMPARISON OF THREE DIFFERENT AUXILIARY POWER SYSTEMS

Power systems are designed to furnish an average load of 1300 watts to supply the power needs of a direct mission to the Moon.

Solar Photovoltaic Power System	1146 lbs
Isotope Dynamic Power System	1268 lbs
Stand-by Redundant Fuel Cell System	1309 lbs

The weight difference calculated seems to be considerable, however, the fuel cell system has the highest estimated reliability and is most fully developed. The solar cell system needs additional mechanical design work if used for landing on the Moon. The solar cells also need additional equipment to orient them toward the Sun independently from the capsule attitude. This may increase the weight of the power system up to 200 pounds.

The isotope-dynamic power system has no built-in redundancy and, therefore, has lower reliability than the other two systems. A redundant turbine generator-heat exchanger system may increase the weight approximately 50 to 100 pounds, considering that electrical switching and safety-valve connections may also be needed.

The power need of the stabilization and control system may be reduced subtantially, based on the recent experience of the Mercury Astronauts. This would reduce the weight of the fuel cell system the most, and that of the solar cell system, the least.

Considering all these factors, the fuel cell system is still the heaviest, and the solar cell system the lightest.

7.6 SELECTION OF STORAGE BATTERIES

The power requirements for the Apollo vehicle were established on the basis of the stripped-down requirements established for the Apollo project. Based on the available load profile, a primary system rated at 1300 watts or less with a storage battery system, will handle any peak loads. It is the intention to use the storage battery selected for reentry also as the auxilliary power source for peak loads. Limiting the peak power requirements on the primary power source without considerable overload requirements, results in reduced size and weight with resultant higher reliability.

The storage battery system can handle considerable overload for a short time and is designed with an emergency back-up battery. The back-up battery is sufficient in size to handle emergency conditions and to provide secondary power for return mission in case the main battery fails. Selection of reduced-size back-up battery reduces weight without reduction in reliability.

7.7 RELIABILITY TESTING

The power system reliability has to be demonstrated before the mission would be attempted. For reliability testing, a number of power systems have to be made available so that simultaneous testing can be performed. It is estimated that, at least, six or eight power systems have to be available to establish the reliability at a reasonably confidence level. The availability of the isotope fuel is limited, and it may have to be simulated with electric heating. This may not be satisfactory for some reliability people.

The solar cells are extremely expensive and one power system's cost is about \$2x10^6. The cost may prohibit the building of six or eight power supplies for testing only. On the other hand, a reduced size system can be tested for reliability at a satisfactory confidence level.

7.8 POWER DISTRIBUTION SYSTEM CONSIDERATIONS

The power distribution system on the LEM version of Apollo is a combination of AC and DC power. The AC distribution system is 400 Cycles at 115/200 volts 3 phase.

Our investigations show that the new design proposed has considerably less motor loads and the justification of a major central AC distribution system is not clear cut. There may be a considerable weight saving and flexibility in design using individual inverters for most of the motors in the ECS system and the fuel cell system. Elimination of output transformers reduces the inverter weight. Use of individual inverters for each motor increases system reliability and reduces radiated radio noise.

A small amount of AC power will be provided at low voltage. This power will be used for the battery chargers, for guidance and navigation, and stabilization and control. The total AC power need does not exceed 250 watts.

A 3 phase 400 cycle low voltage quasi sinusoidal AC power will be provided. Two inverters will be used and only one inverter will be operating at a time.

The total power requirements are approximately 395 KW hours. The required fuel is 1.3 times this value in pounds or 515 pounds, or approximately 58 pounds of H_2 and 460 pounds of O_2.

Considering the possibility of using common storage tanks with the propellant system, a reduced fuel tank weight can be achieved. A separate fuel storage for the fuel cells cannot be eliminated completely for safety reasons, but the size of such tanks can be reduced to store only fuel for emergency use which should be enough for a safe return trip.

8.0 ENVIRONMENTAL CONTROL

Basic to the use of cryogenic propellants for space travel is the implication that such fluids can be stored for long periods without incurring serious weight penalties. The propellants must be maintained as liquids in order to require a minimum storage volume and must be maintained at low temperature in order to avoid high storage pressure with consequent high storage vessel weight. Storage vessel configuration must result in a maximum density of loading of the spacecraft in order to keep its volume to a minimum and its stability at a maximum.

Insulation of cryogenic storage vessels must be highly effective, and their supports and accessory connections must be designed to prevent appreciable heat leak. Heat leak to the stored fluid from any source must be minimized for spacecraft applications because waste overboard requires carrying unnecessary propellant which is directly deductible from payload weight. The application of moderately effective insulation increases weight and volume of spacecraft and correspondingly decreases payload.

The insulation and support structure must be arranged to effectively avoid high heat leak rates or condensation of water vapor or air during ground hold and boost. Moisture or ice formation decreases insulation effectiveness for a considerable time after boost, and its weight directly decreases boost payload and may affect spacecraft payload to an important degree.

8.1 CONDITIONS

The cryogenic storage vessel and insulation will be exposed to a wide range of environmental conditions during ground hold, boost, space trajectory, lunar orbit, and lunar stay. The tank insulation and suspension system must be capable of withstanding these environments and of maintaining the proper thermodynamic states of the stored propellants. Conditions within the propellant tanks must be such that the propellants may be delivered to the engines at the proper temperatures and pressures for efficient utilization.

During ground hold, the insulation and suspension system will be exposed to atmospheric pressure, to atmospheric water vapor, and to the "permanent" gases of the atmosphere, practically all of which condense at temperatures above the required storage temperature of liquid hydrogen propellant. This environment must not destroy the insulation effectiveness by crushing or by condensation or freezing of its constituents on the surface or interior of the insulating blanket.

The boost phase will result in high acceleration force, mechanical vibration of the spacecraft, acoustic vibration, and aerodynamic heating. It may be assumed that these forces, except acceleration forces, will be primarily accepted by a shell exterior to the storage vessels from which tney are supported. The effect on the insulation, piping, and supports must be accounted for to be certain that no damage can result during boost. Acceleration forces will impose additional stresses on vessel walls and supports and insulation mounting.

In Earth orbit and trans-lunar trajectory, an important alteration in environment is the absence of atmospheric pressure and of atmospheric gases. The spacecraft also is in a "zero-gravity" field, and the primary heat imputs come from solar radiation, and from the albedo of the Earth and

Moon when the spacecraft is near these bodies. The traverse of the Van Allen belt will result in exposure to appreciable nuclear radiation, but only for a short period (less than an hour). Exposure to solar flare activity would result in extreme nuclear radiation exposure, but, presumably, this condition can be avoided by judicious scheduling of a flight. Exposure to nuclear radiation can result in heating of propellants, in addition to its effect on personnel and materials. The extent of this effect depends on the concentration of gamma particles and the corresponding cross-section of the fluid.

The further alteration of environmental conditions during lunar orbit and lunar stay depends on the altitude and orientation of the orbit and on whether the stay is during lunar day or night. The ambient pressure remains low during this period and there are no atmospheric gases present. The Moon's gravitational effect will be present from de-orbit until lunar orbit after takeoff. Acceleration forces will be present during lunar landing and takeoff. During lunar stay, the spacecraft will be exposed to direct solar radiation, reflection from the lunar surface, and re-radiation from the surface at temperatures approaching 250°F. if the stay is during lunar day. During lunar night, exposure would be to the surface at temperatures approaching -250°F. with no exposure to direct solar radiation or albedo.

8.2 HEAT LEAK TO PROPELLANT TANKS

The primary problem in protecting propellant storage from heat leak occurs during the ground hold. This is the shortest phase of the mission but involves convection and condensation in addition to radiation and conduction as important modes of heat transfer. Condensation of ambient water vaper will take place on any surface which is below the dew point temperature of the air, and condensation of air, on any surface colder than the condensing temperature of air. The storage vessel sidewall area is the largest concerned, but condensation on supports and piping connections can contribute unacceptably large losses. Condensation not only causes a high rate of heat leak, but also results in added boost weight and decreased effectiveness of insulation until the resulting liquid or solid is boiled off under the low pressure conditions of the space environment. Heat leak may enter directly through the wall insulation and through structure or fittings used to position the insulation; through tank supports by convection, radiation, and conduction as well as by condensation; and through piping and accessory connections by these same mechanisms.

The action of aerodynamic forces during boost results in relatively high heat flux on exposed surfaces. The period of time involved, however, is approximately forty seconds, and the area of exposure is the spacecraft skin, rather than the tank and support surfaces. This problem is, therefore, no more serious than protection during ground hold because the time and the total heat accepted by the structure are small. The rise in temperature of the structure will be limited because of its heat capacity, and the conduction path through the tank supports will be of maximum length.

Heating in space and during lunar stay will be a function of the temperature attained by the outer skin of the spacecraft due to the equilibrium of incident high temperature radiation and cooling by radiation to space. The estimated skin temperature as a function of time, after leaving Earth orbit, is shown in Figure 8.1. Similar studies concerning the Apollo spacecraft have indicated an appreciable temperature differential circumferentially due to partial exposure to space and partial exposure to solar

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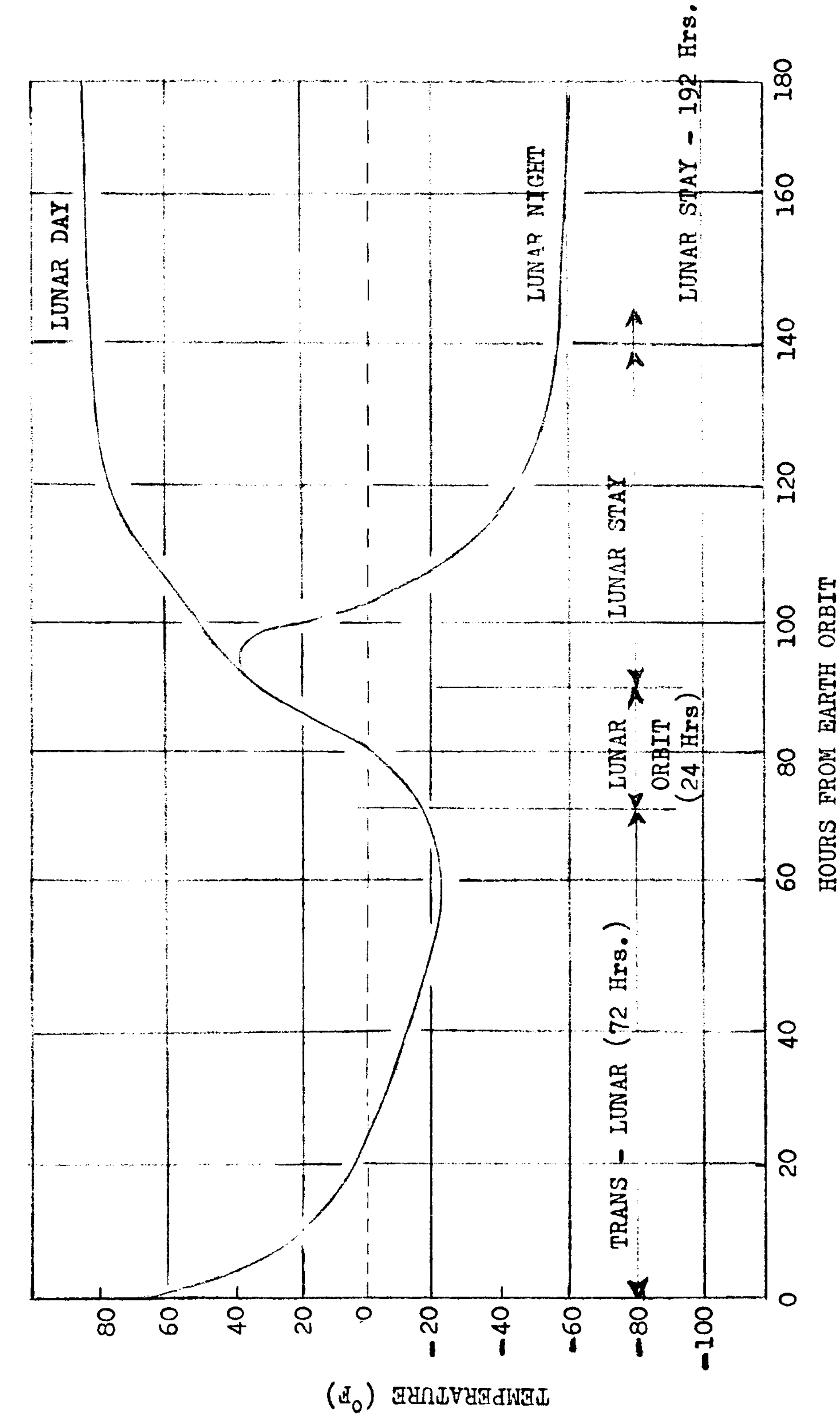

Figure 8.1 TEMPERATURE OF S/M SHELL - INSIDE ALUMINUM HONEYCOMB

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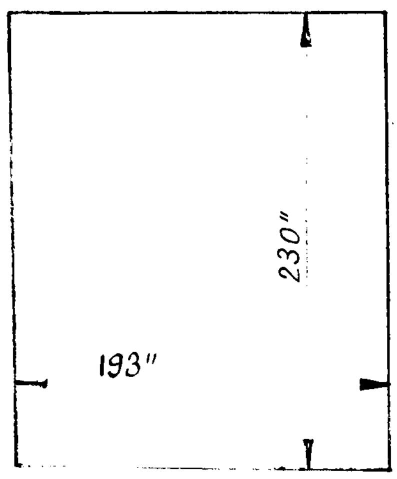
radiation and other heat inputs. Slow rotation of the craft (in the order of 1 rpm) about an axis normal to the major direction of heat input appears feasible; in fact, more so than preventing rotation or maintaining constant orientation.

The greatest requirement for high effectiveness of insulation is that for residence on the Moon during lunar day, particularly at the time of lunar noon. This is the longest period of exposure to high incident heat flux, and during this time, no rotation of the spacecraft is possible to equalize skin temperature. The propellant tank insulation, however, will equalize temperature in its outer layers because the circumferential conductance is many times its radial conductance. The following computations indicate the expected mean skin temperature during residence on the Moon's surface during lunar day.

<u>Calculation of Skin Temperature</u>:

Assume Rt. Circular Cylinder, as indicated



1. Solar radiation to cylindrical wall - 443 Btu/ft^2-hr (normal)

   Equivalent radiation to exposed semi-cylinder + KDL =

   $= 0.3 \times \frac{443 \times 230 \times 194}{144} = 40{,}500$ Btu/hr.

2. Radiation from surface of moon to semi-cylinder, as above:

   Form factor = 0.3 (assumed) α Moon = Emoon = 1.0

   T surface = 250°F = 700°R   E surface = 0.3

   $R_m = 0.173 \times 0.3 \times F\ (7.0)^4 - \frac{T_1}{100}\ 4$

   2(a) - Refractory effect of lunar surface = KDL (α) F=40,500 x 0.3 = 12,200 Btu/hr

3. Radiation to space:

   $R_s = 0.173 \times 0.8 \times \left(\frac{T_1}{100}\right)^4$ (disregard temperature of space above absolute zero)

4. Assume that temperature equalizes around circumference of cylinder. Lateral area = $\frac{\pi \times 230 \times 194}{144} = 970$ sq. ft., use 1000 sq. ft.

   Radiation from lunar surface + solar radiation - heat entering vehicle - radiation to space = 0

5. Heat Balance

   $$40{,}500 + 0.173 \times 0.09 \times \left[2401 - \left(\frac{T_1}{100}\right)^4\right] \frac{\times \pi \times 230 \times 194}{144 \times 2}$$
   $$- 660 - 0.14 \times \left(\frac{T_1}{100}\right)^4 \frac{\times \pi \times 230 \times 194}{144} = 0$$

   $T_1 = 510$ R = 50 F.

Computation on Apollo go as high as 160°F. Skin temperature and ε of paint is uncertain because emissivity of paint may vary. 80°F is a conservative estimate.

If the form factor for lunar radiation were 0.5 instead of 0.3; and, if the absorptivity of paint were 0.8 (equal to its emissivity), the computed surface temperature would increase.

1. Solar radiation = $40{,}500 \times \frac{0.8}{0.3}$

2. Radiation from lunar surface = $0.173 \times 0.5 \times 0.8 \times \left[2401 \left(\frac{T_1}{100}\right)^4\right] (500)$

2(A). Refractory effect from lunar surface = $40{,}500 \times \frac{0.8}{0.3} (0.5)$

3. Radiation to space (same as before) = $2140\left(\frac{T_1}{100}\right)^4$

4. Equating to zero, as above, results in T_1 = 140 F, maximum expected skin temperature.

8.3 HEAT CONTROL PROVISIONS

Three types of heat control provisions are proposed:

1. Loading of propellants in the "super cooled" state.
2. Optimum application of the usage cycle of propellants and maximum use of stored fluids for other requirements (environmental and auxiliary power needs).
3. Proper application of "super-insulation" and design of tank mounting for minimum heat leak.

The duration of a proposed expedition of fourteen days is short enough that sub-cooling of propellants may be applied to alleviate insulation requirements. The primary application is to the second stage, since the first stage propellant will be used during lunar trajectory, orbit, and landing. There is a minimum temperature of propellants for engine operation, and it may be necessary to use the second stage engines if the expedition should be aborted after lunar landing operations have begun, so sub-cooling may be applied only to such an extent that the temperatures of propellants will have increased to the minimum required for engine ignition. One procedure for sub-cooling is by pumpdown of the vapor space in a storage vessel, causing evaporation to take place from the stored liquid with resulting temperature decrease. This procedure will take a number of days for the quantity of cryogen required, so the pumpdown would make use of a separate storage as part of Ground Support Equipment. Transfer to the spacecraft tanks would take place during final countdown as a rapid transfer process.

Sub-cooling of LH_2 to $33^{\circ}R$(6.5 psia saturation pressure), and of LO_2 to $155^{\circ}R$(10 psia saturation pressure) will result in corresponding temperatures at lunar arrival of $36^{\circ}R$ and $170^{\circ}R$ which are at the lower limit of engine operating temperatures. This permits the propellant temperatures to rise during lunar stay through the entire allowable operating range.

The cycle of propellant usage will assist in avoiding loss of propellant through boil-off, if tank pressurization is properly controlled. Required usage occurs at mid-course correction in trans-lunar trajectory at lunar landing, lunar take-off and orbit, and mid-course correction in trans-earth trajectory. Use of pressurization gas at the optimum temperature for minimum gas requirement and the minimum pressure necessary for engine operation, at each engine use, will result in decrease of propellant saturation pressure and temperature after each such use. The cool-down of pressurizing gas will result in a decrease in pressure in the storage tank ullage, followed by evaporation of stored fluid and decrease in its temperature. The extent of this effect varies with the propellant quantity used. Use of stored fluids for environmental control and power generation will have a similar effect on decreasing the likelihood of propellant boil-off, in addition to a direct decrease in payload weight due to avoiding separate tankage for such requirements.

After accomplishment of lunar landing, the propellant tanks for the landing engines will contain, as a minimum, no liquid but the tank volume full of super-cooled vapor, or as a maximum, about 5% remaining liquid propellant. These materials can replace power and environmental requirements to some degree during lunar stay, and their evaporation and venting can provide a thermal barrier to incident heat during this period. The available materials and heat capacity should be utilized to the greatest practicable extent.

The heat which may be accepted by the vapor in the landing tanks may be approximated as follows:

Initial state - Saturated at $40.3^{\circ}R$

Final State - Superheated to $540^{\circ}R$ ($70^{\circ}F$)

The heat content at any time is equal to the product of mass heated, its specific heat, and the temperature increment.

$$dQ = W\,C_p\,dT$$

From the Perfect Gas Law:

$$PV = WRT$$

At constant pressure and volume:

$$WRT = K = W_1RT_1$$

$$W = \frac{W_1T_1}{T}$$

Substituting:

$$\int_0^Q dQ = Cp \int_{T_1}^{T_2} WdT = C_p\,W_1\,T_1 \int_{40}^{540} \frac{dT}{T}$$

$$Q = C_p\,W_1\,T_1\,\ln\frac{540}{40}$$

$$= 3.2 \times 337 \times 40.3 \times 2.6 = 113{,}000 \text{ Btu}$$

which may be accepted by the vapor in the landing tank acting as a heat barrier to protect the takeoff tanks.

The application of "super-insulation" and design of tank mountings are covered in the following sections.

8.4 INSULATION REQUIREMENTS AND TANK SUPPORTS

The insulation and tank support requirements are discussed in detail in "Lunar Mission System Studies - Interim Report," SID 62-1189.

8.5 CONTROL OF PROPELLANT LOSSES

Cooldown of propellant tanks and piping will be provided primarily by gaseous nitrogen followed by liquid nitrogen purge. The relatively small additional boiloff required to cool to liquid hydrogen temperature can be compensated by topping. Tank insulation must be highly effective and efficiently sealed to provide a high vacuum and to exclude moisture, otherwise maintenance of sub-cooled liquid cryogen would be impossible to maintain and topping and venting requirements would be excessive. A requirement for high topping rates would be impractical also from the standpoint of ground held requirements. It is desirable to have the capability for reasonably long ground held after disconnect of ground support equipment to permit adequate time for final checkout before lift-off.

Propellant losses after lift-off are discussed in the preceding sections.

8.6 INSULATION FABRICATION

As discussed in "Lunar Mission System Studies - Interim Report," SID 62-1189, a program for the study of super-insulation installation techniques is now in progress. The present indication is that a suitable sealing film with stand-off columns can be installed at a weight penalty about equal to that of the super-insulation. In the case of LH_2 storage, this amounts to a total of 0.20 to 0.25 lbs/sq. ft.

The studies to date have been based on the application of $\frac{1}{2}$ inch of super-insulation to the outside surface of the cryogenic storage vessel, and the protection of this blanket with a vacuum - tight external diaphragm. The diagphragm will be held away from the insulation by columns $1\frac{1}{2}$ inches high and about 12 inches on centers; the actual spacing depending upon the geometry of the storage vessel being insulated.

In operation, a vacuum pump will be connected to the diaphragm and the space between the diaphragm and storage vessel shall (including the super-insulation) evacuated to 100 microns of mercury, or less. The system can then be checked for tightness and structural integrity. Cooldown and fill of the storage vessel will then "cryopump" the insulation to less than 10^{-4} min. Hg, and the insulation will be fully effective.

8.7 SKIRT PROTECTION FROM LANDING ENGINE BLAST

The engines for Lunar Landing are recessed in the skirt of the landing module. They are arranged to gimbal 18^{o} in one plane and 5^{o} in the 90^{o} plane, in order to furnish maneuverability and safety in the event of failure of one engine during landing operations.

The engines operate at 300 psia and 5000^{o}R in the combustion chamber and an expansion ratio of 40 to 1. Engine operation may be required for a maximum of 300 seconds on, followed by 200 seconds off, although the "on cycle" will normally be considerably shorter than this.

The jet of combustion gas consists of H_2O, exhausting to a pressure of 10^{-12}mm Hg absolute. Since the jet exhausts to a very low ambient pressure, the spread and the acceleration will be very rapid. If no structure were present, the jet would spread to about 100° from the engine longitudinal centerline. When the engine is gimbaled 18°, the jet impings on the structure about 30° from its outside surface. The portion which impinges, however, is at high Mach number (above 20) and at low temperature, due to the rapid expansion of the gas stream.

Former studies by Rocketdyne division for the LEM indicate that an engine operating under the conditions outlined above will emit a jet at about 1500°R for 5 diameters along the nozzle centerline. At about 10° from the centerline, the temperature will be 1100°R and the threw 10 diameters. At 45°, the temperature will be less than 400°R, the Mach number about 12, and the threw about 60 diameters. Since the portion of the flame which impings on the skirt is at low temperature, the heating effect will come from radiation from the equivalent mass of the jet which is at a relatively high temperature. It is assumed that the equivalent heating effect may be represented by a spherical flame 10 outlet diameters in size, at a temperature of 1000R, and centered 16 diameters from the nozzle outlet plane.

The maximum heating effect, by radiation, on the closest portion of the skirt is less than 2 Btu/sq ft/hr. If the maximum period of engine firing is 300 seconds, the maximum heating of a "black" skirt surface would be $2 \times \frac{300}{3600} - 0.17$ Btu. If the skirt material has a mass of 0.2 lbs/sq. ft. and a specific heat of 0.05 Btu/lb. F, the temperature of the skin would rise less than 20° in 300 seconds.

The heating effect of the jet is low primarily because of the rapid expansion in vacuum surroundings. The energy of the exhaust is rapidly used in acceleration of the gas stream, with resulting dip in temperature. In addition, the emissivity of H_2O gas is low even under reasonably high pressure. In this case, the emissivity is estimated to be less than 0.01. Convection is not a factor because of the low temperature of the impinging gas, due to the rapid gas expansion discussed above.

9.0 STRUCTURES

In addition to providing support for the weight estimates on numerous configurations of the direct mission cryogenic vehicle, a detailed structural analysis was conducted on the base point configuration of Reference (1). The results of this comprehensive study are given in Reference (2). It had been shown previously that, with a command module weight of 10,000 pounds, the direct mission could be achieved with a C-5 booster, provided that a high-energy cryogenic propellant lunar landing and launch system was used. This material, a summary of Reference (2), defines the current ground rules and criteria for this study.

9.1 DESIGN CRITERIA

The criteria used for the structural analysis of the direct mission vehicle is consistent with current Apollo criteria with the following exceptions. Booster acceleration histories and skin temperatures used for this study are shown in Figures 4.3 through 4.6 and 9.1 and 9.2. The maximum dynamic pressure was assumed to be 800 lbs/sq ft, which could be combined with a maximum angle of incidence, $\alpha = 10°$, as shown in Figure 9.3. The propulsion system and tanks were designed consistent with the pressure, load factors, and temperatures shown in Table 9-I.

Each engine was assumed to have a maximum thrust of 15,000 pounds and was capable of gimbaling within a semi-cone angle of 5 degrees. The landing gear was designed to the conditions summarized in Table 9-II. The vehicle and vulnerable equipment was protected against meteoroid penetration to a degree compatible with the over-all vehicle requirements, assuming a

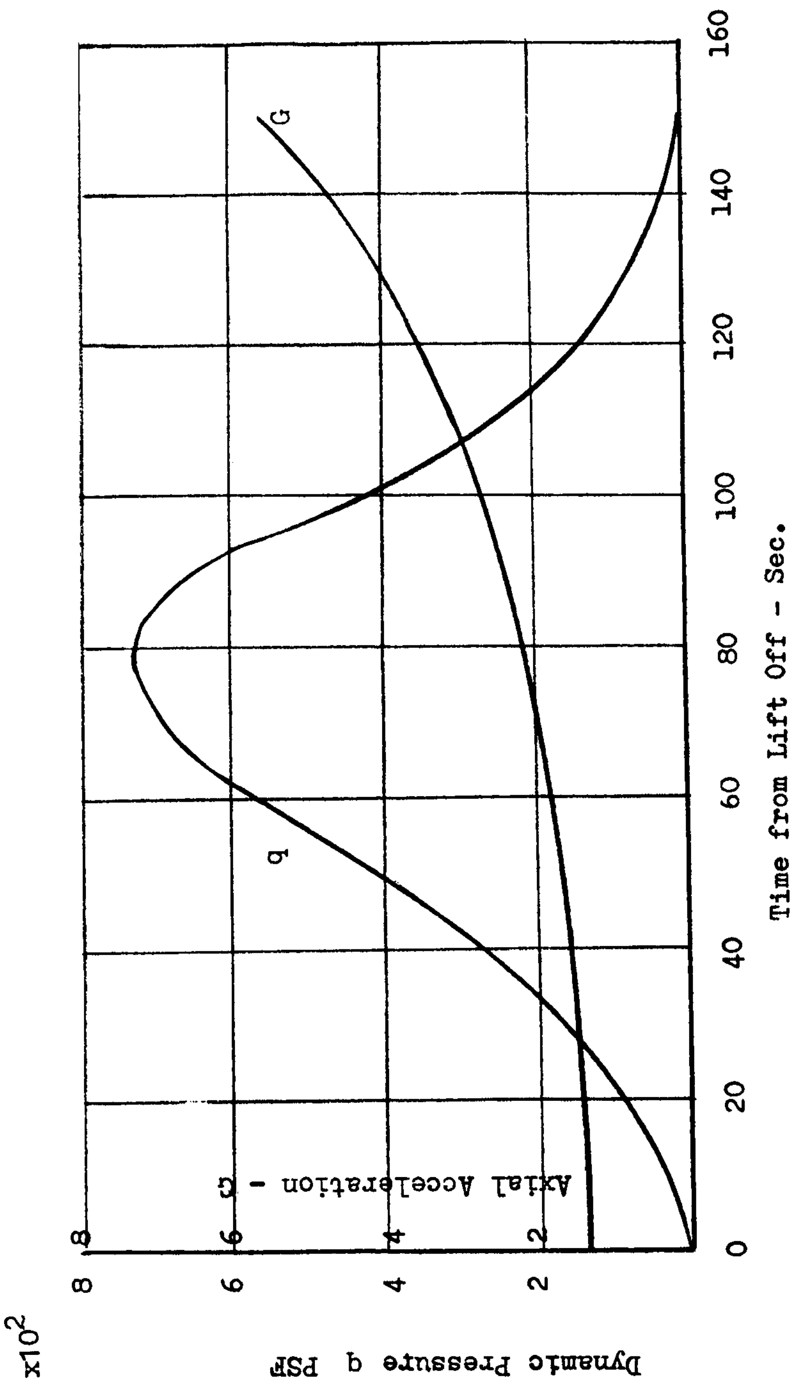


Figure 9.1 AXIAL ACCELERATION & DYNAMIC PRESSURE HISTORIES - C-5 BOOSTER

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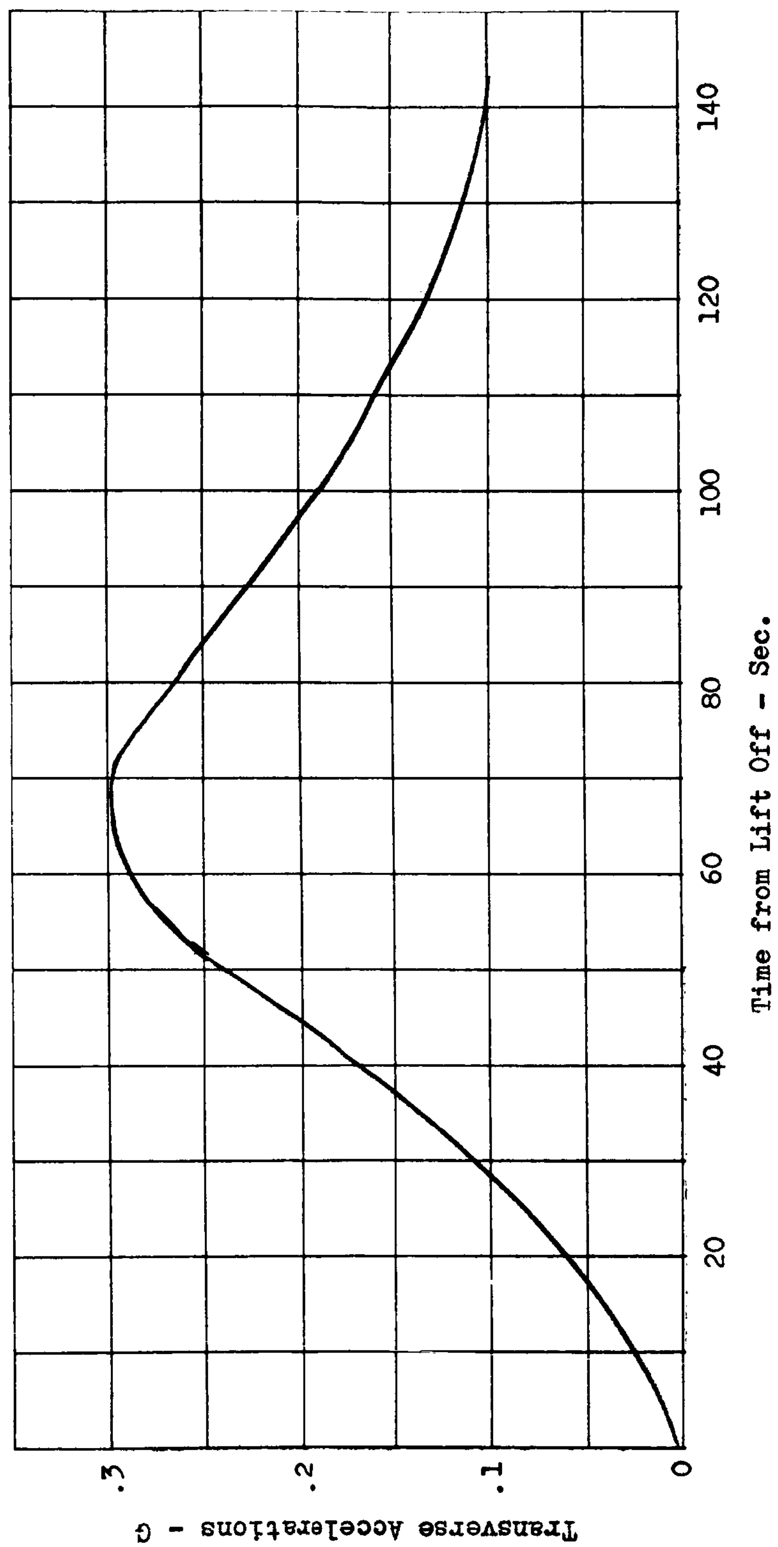

Figure 9.2 TRANSVERSE ACCELERATION HISTORY - C5 BOOSTER

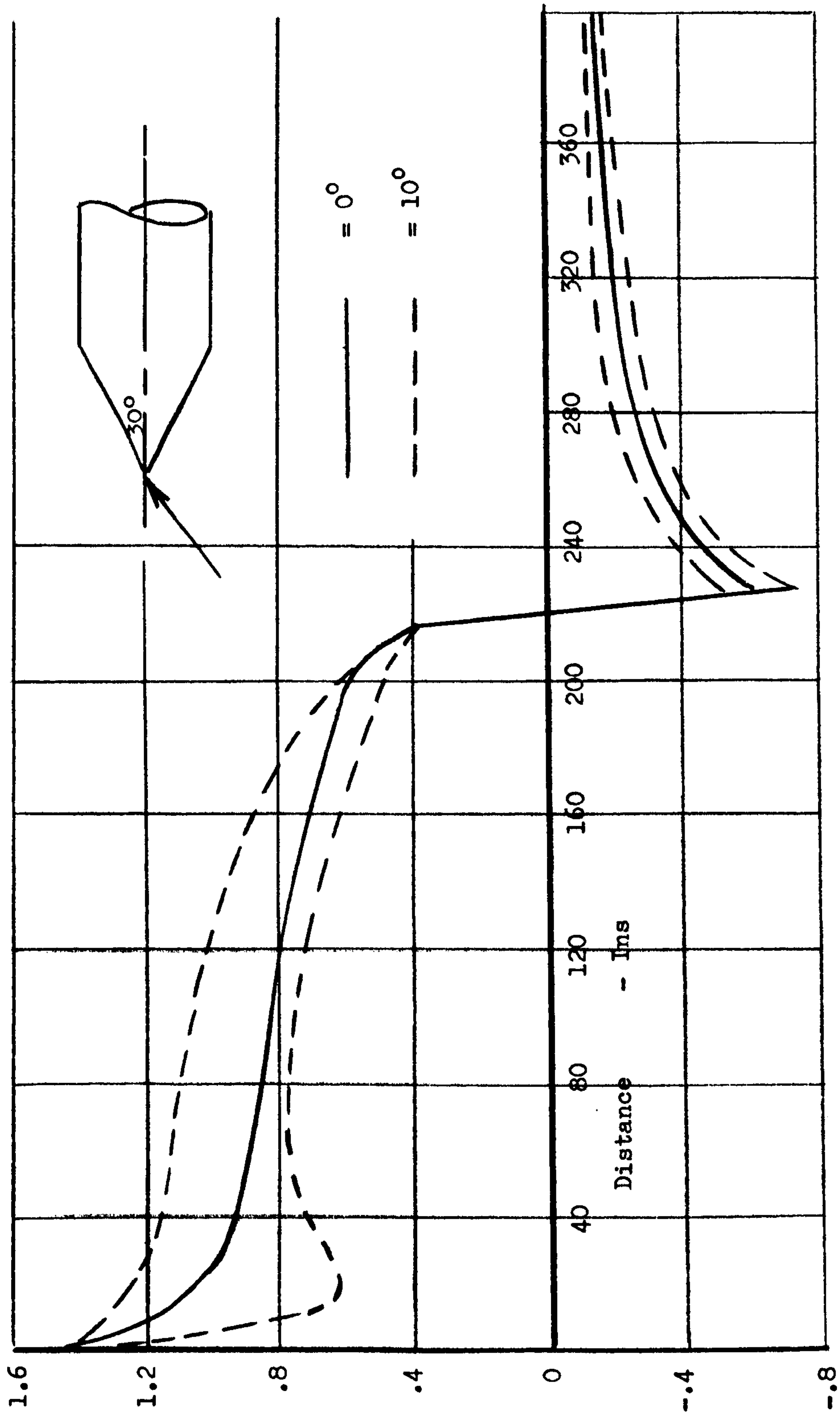

Figure 9.3 PRESSURE DISTRIBUTION OVER THE STUDY VEHICLE

Whipple 61 meteoroid flux and ballistic coeffiecient of penetration given in Reference 3.

TABLE 9-I

PROPULSION SYSTEM AND TANK CRITERIA

System	Max. System Pressure	Ultimate Load Factor	Temperature (F)
LO_2	70 / 48	1.5	-360
LH_2	51 / 40	1.5	-430
Other high-pressure tanks		2.0	

9.2 EXTERIOR STRUCTURE

9.2.1 Material Selection

Material selection was predicated on the skin temperature histories illustrated in Figures 4.3 through 4.6. Briefly summarized, these figures indicate:

1. Stage I - Maximum predicted wall temperature is 272°F for the uninsulated shell.

2. Stage II - Maximum predicted wall temperature is 798°F for the uninsulated wall and 290°F for the insulated wall.

In view of the above, aluminum without insulation will be employed in the construction of Stage I, but insulation would be required to permit the use of aluminum for Stage II since the aluminum would be structurally inadequate at 798°F. However, all known lightweight insulators char upon ablation and present a black body detrimental to the heat rejection system. Therefore, materials that could withstand high temperatures such as titanium, PH 15-7Mo Steel, and Rene 41 were investigated with due consideration of the significantly greater wall temperatures these materials would experience in contrast to that presented in Figures 4.3 through 4.6.

Rene 41 was selected as the most desirable material for Stage II in view of its greater strength reliability at wall temperatures estimated at 1150°F. (See Figures 9.4 and 9.5.)

9.2.2 Design Analysis

Using the weight distribution given in Figures 9.6 and 9.7, and the airloads summarized in Reference 2, bending moment, shear force, and axial load diagrams were developed as a function of the vehicle length. Utilizing these values, the load intensity in the external structure was obtained as a fundamental design parameter, using the combined moment and loads given in Figures 9.8 through 9.11 and Table 9-III.

The design approach utilized for Stage I and Stage II differ in view of the extreme difference in the strength properties of the materials, loading intensity, and exposure time to meteoroids, and are, therefore, discussed separately.

Stage I - The procedure followed in obtaining the optimum method of construction and material was to develop the load intensity parameters in terms of the vehicle diameter and moment, Table 9-III.

Using this value and Figure 9.12, it was found that the most efficient structure would be provided by a honeycomb sandwich material, 2024-T86 (its double-wall structure also is necessary to achieve the desired meteoroid protection).

TABLE 9-II

SUMMARY OF LANDING CRITERIA

Vertical velocity at impact (ft/sec)	10
Horizontal velocity at impact (ft/sec)	5
Ground slope at contact (deg)	$\pm$ 5
Vehicle angle at contact (deg)	$\pm$10
Surface coefficient of friction	0.5
Landing accelerations	$<$ Max boost value

TABLE 9-III

SUMMARY OF COMBINED LOAD INTENSITIES (MAX. $q\alpha$ CASE)

Station	150	200	232	300	400	450
$\Sigma S \times 10^{-3}$	18.45	27.6	29.8	44.7	53.6	75.6
$\Sigma A \times 10^{-3}$	117.1	194	221	227.8	369	369
$\Sigma M \times 10^{-6}$	0.56	1.62	2.86	5.8	11.35	15.09
$\frac{1.5 \Sigma M}{R^2} = N_y$	29	52	79	159	311	425
$\frac{1.5 \Sigma A}{\pi R} = N_y$	289	388	415	417	677	677
ΣN_y (lb/in)	318	440	494	576	988	1102

Having defined the lightest weight construction, the analysis was expanded to determine the details of the exterior structure consistent with

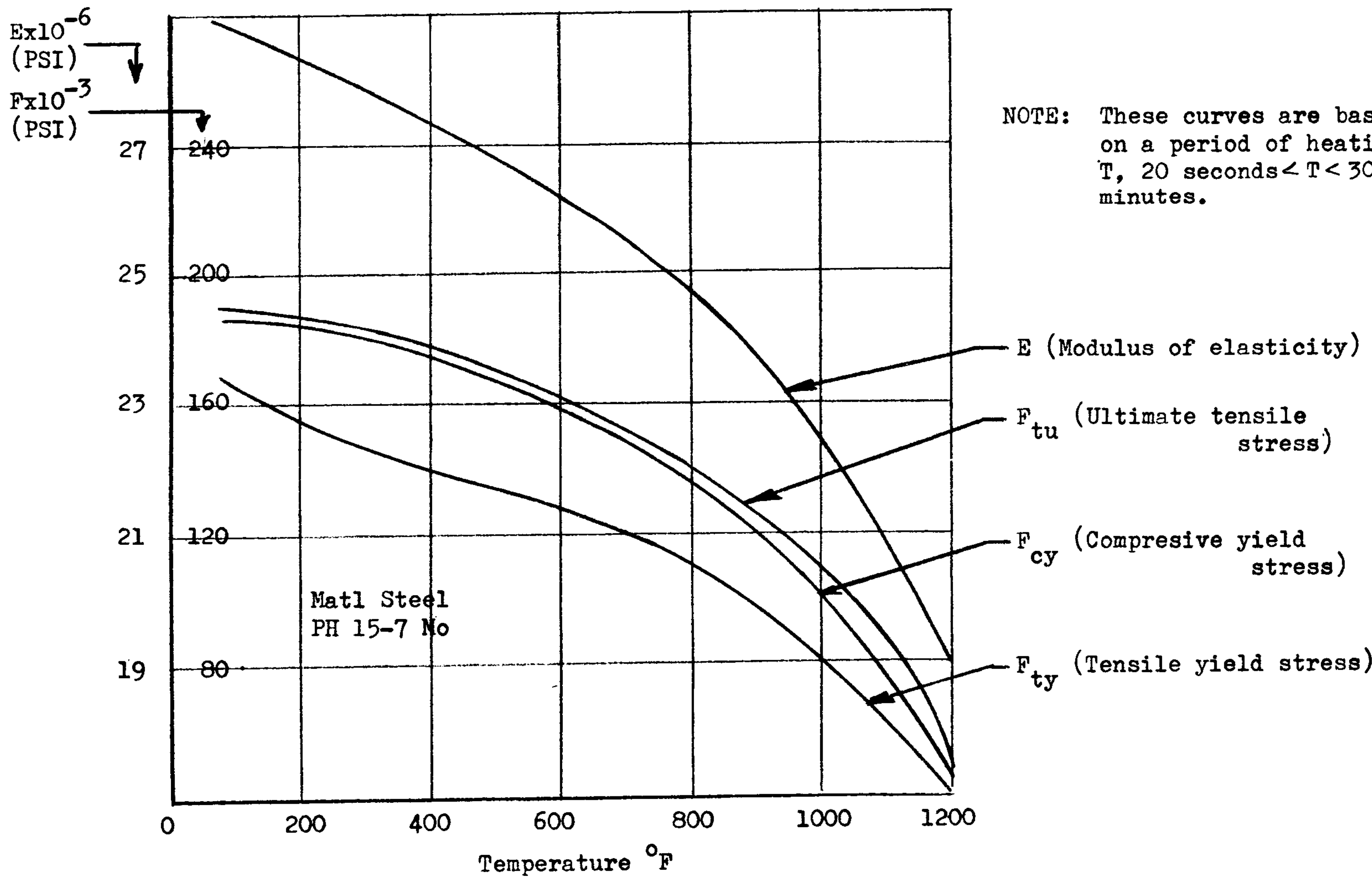

Figure 9.4

PROPERTIES VS TEMPERATURE - STEEL PH 15-7 MO

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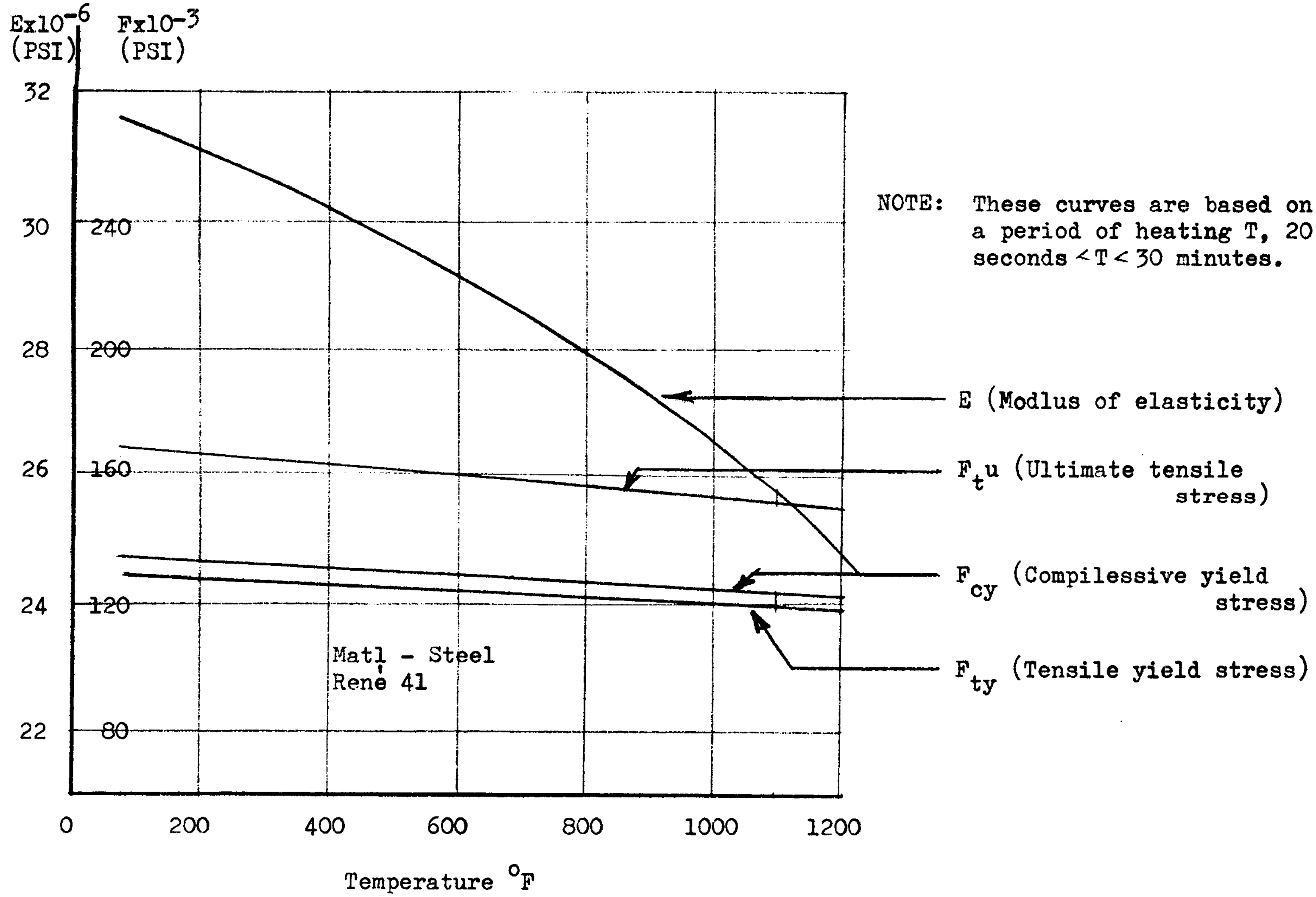

Figure 9.5

PROPERTIES VS TEMPERATURE - STEEL RENÉ 41

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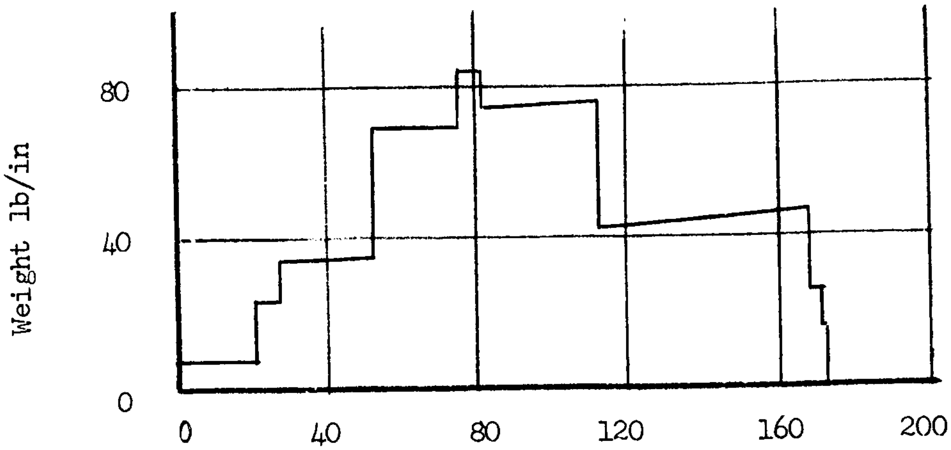

FIGURE 9.6

DEAD WEIGHT DISTRIBUTION LUNAR LANDING STAGE

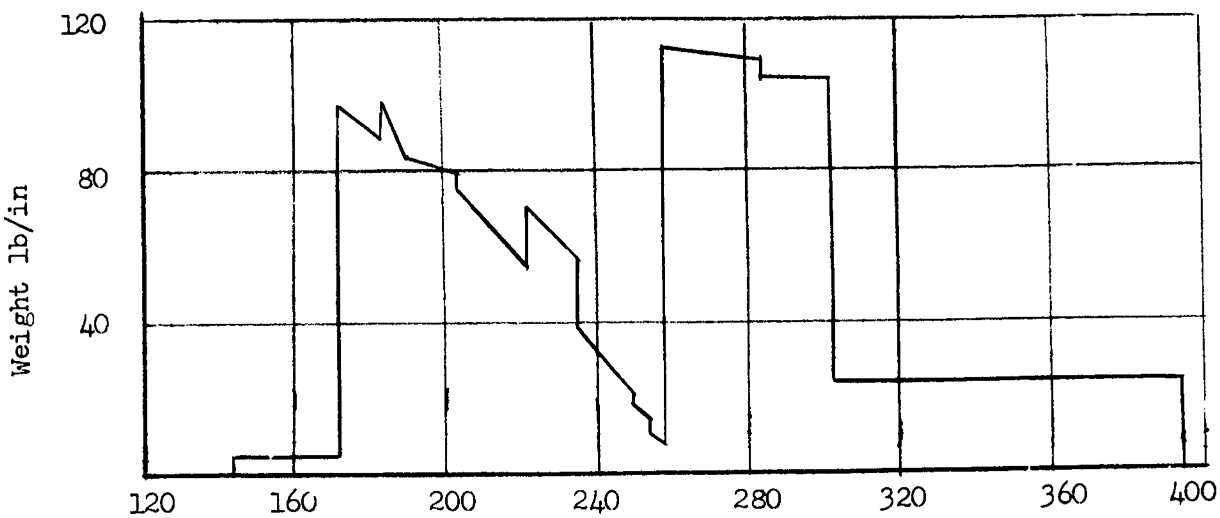

FIGURE 9.7

DEADWEIGHT DISTRIBUTION LUNAR TAKE-OFF

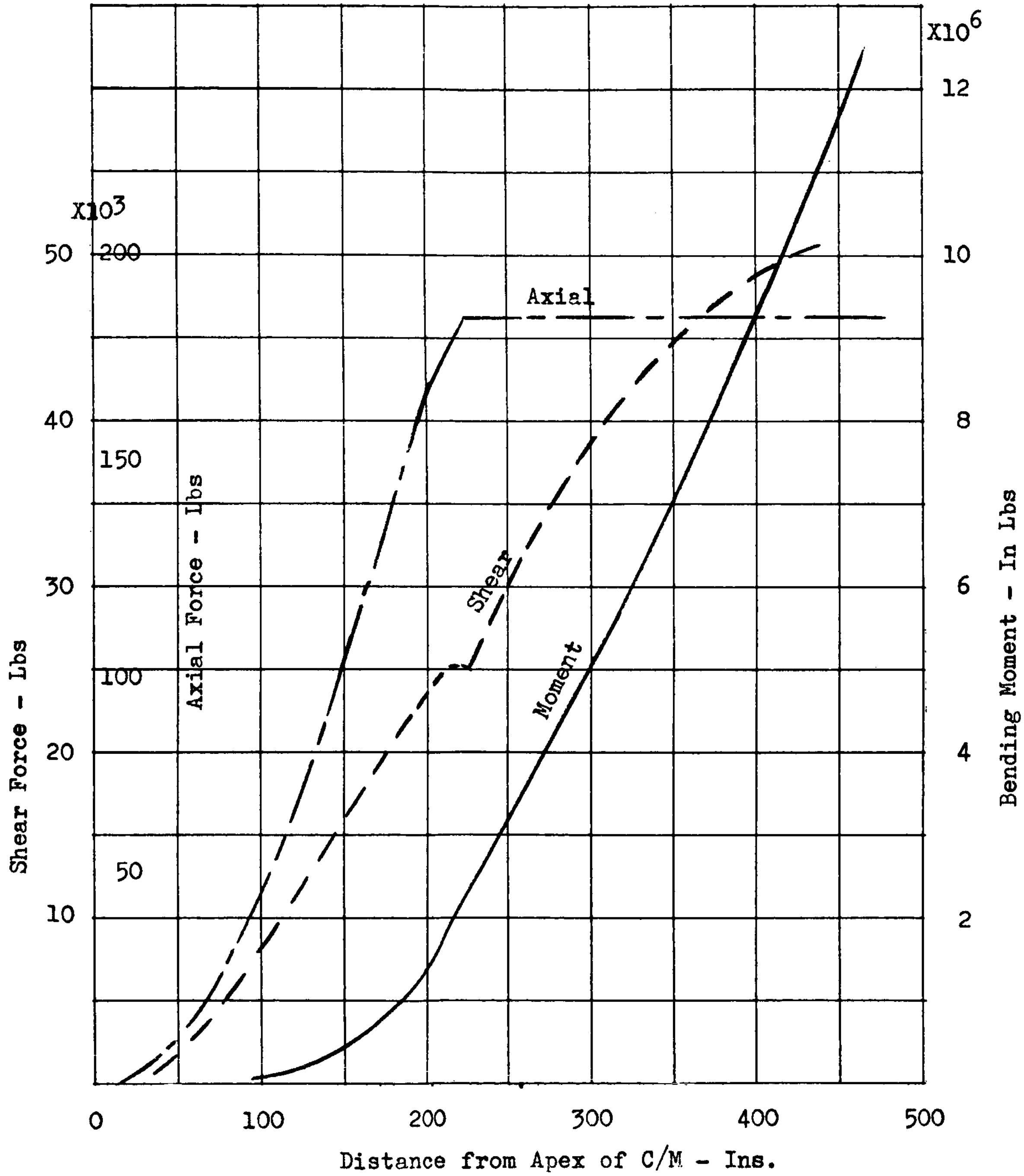

Figure 9.8 SHEAR, AXIAL FORCE & MOMENT DISTRIBUTION DUE TO AERODYNAMIC LOADS -- $q \cdot \alpha$ = 8000 PSF DEG

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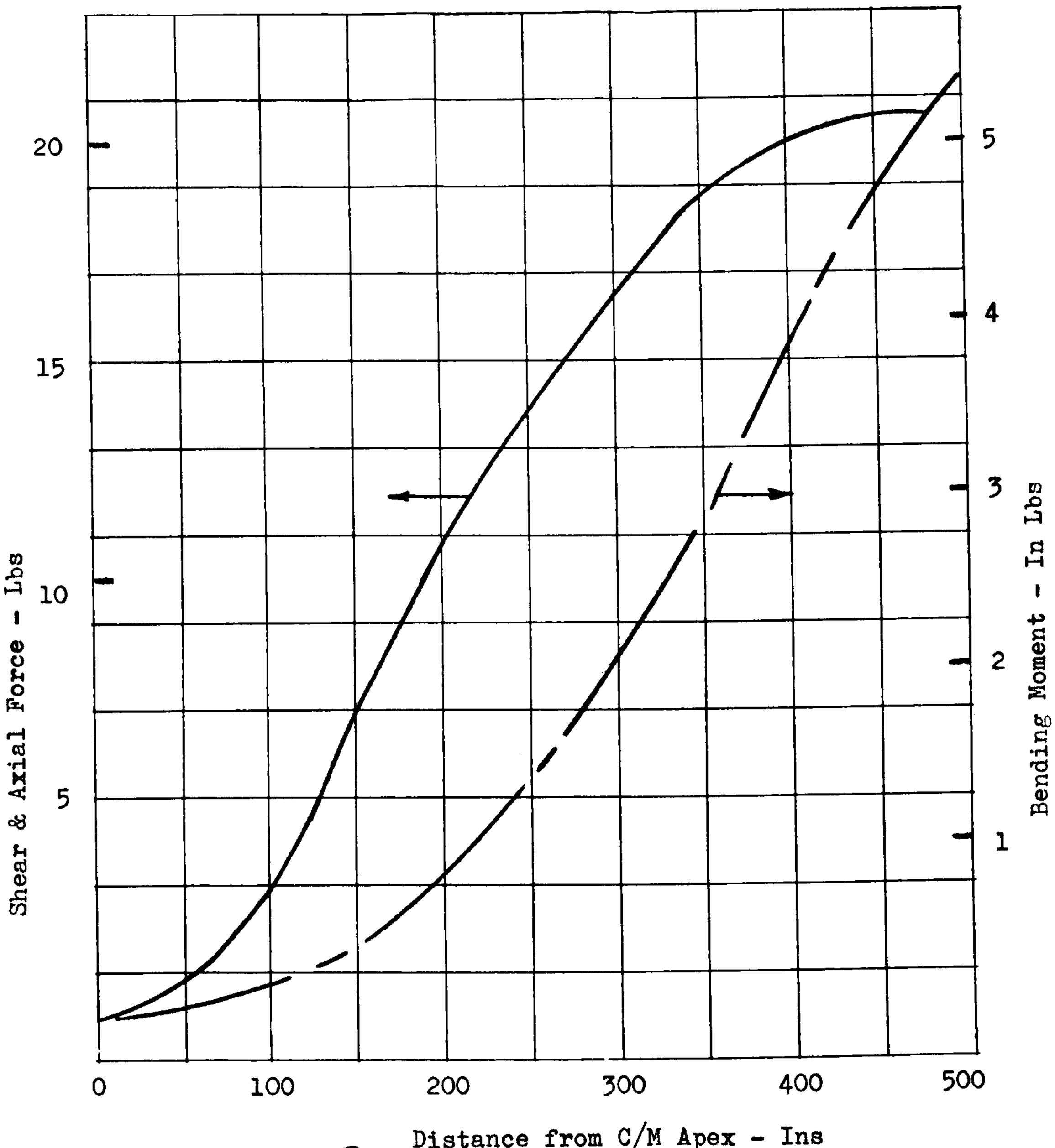

Figure 9.9 SHEAR AXIAL FORCE & BENDING MOMENT DUE TO UNIT ACCELERATIONS

VEHICLE LESS FUEL

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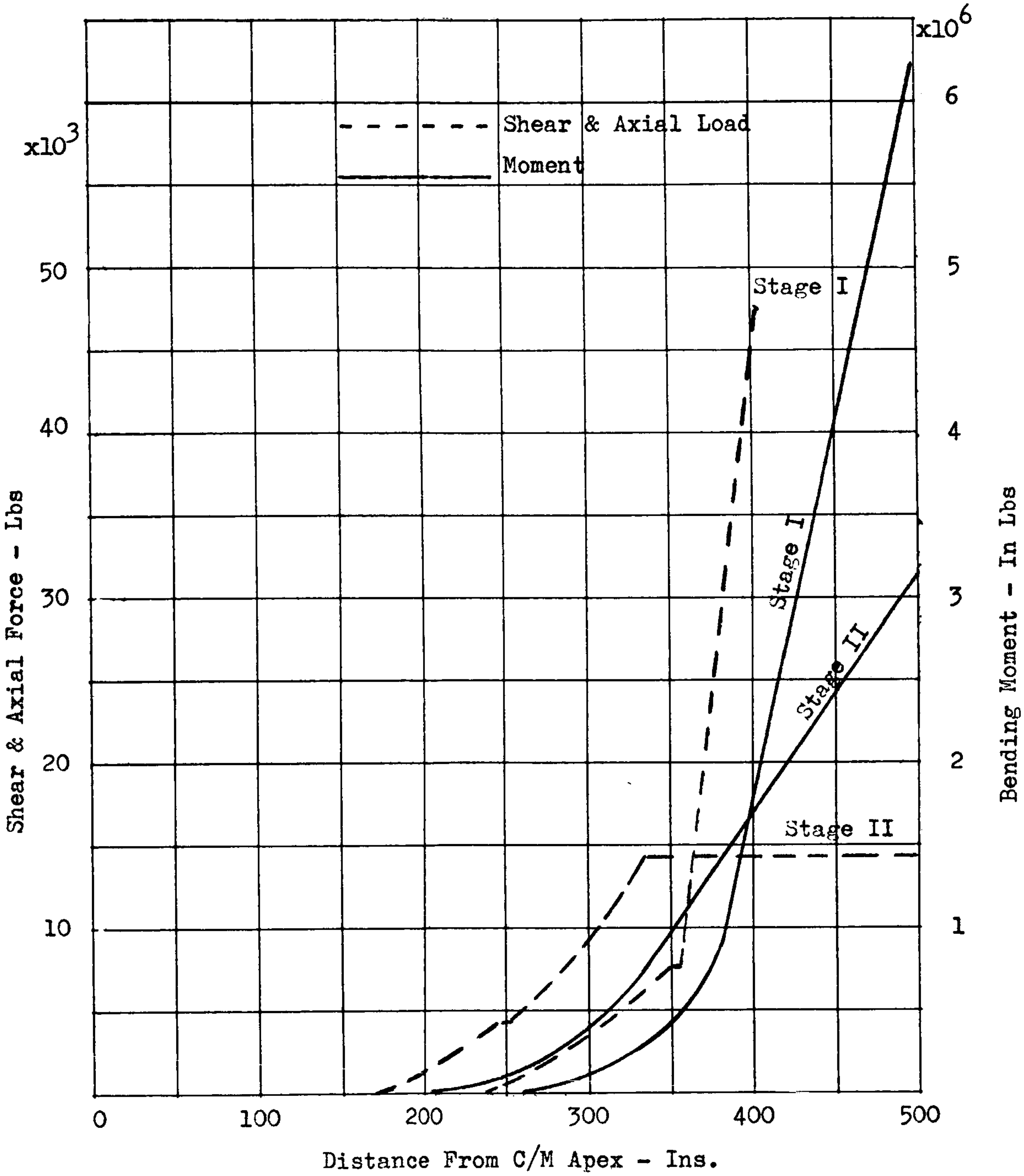

Figure 9.10 SHEAR AXIAL FORCE & BENDING MOMENT DUE TO UNIT ACCELERATIONS

STAGE I & STAGE II PROPELLANT

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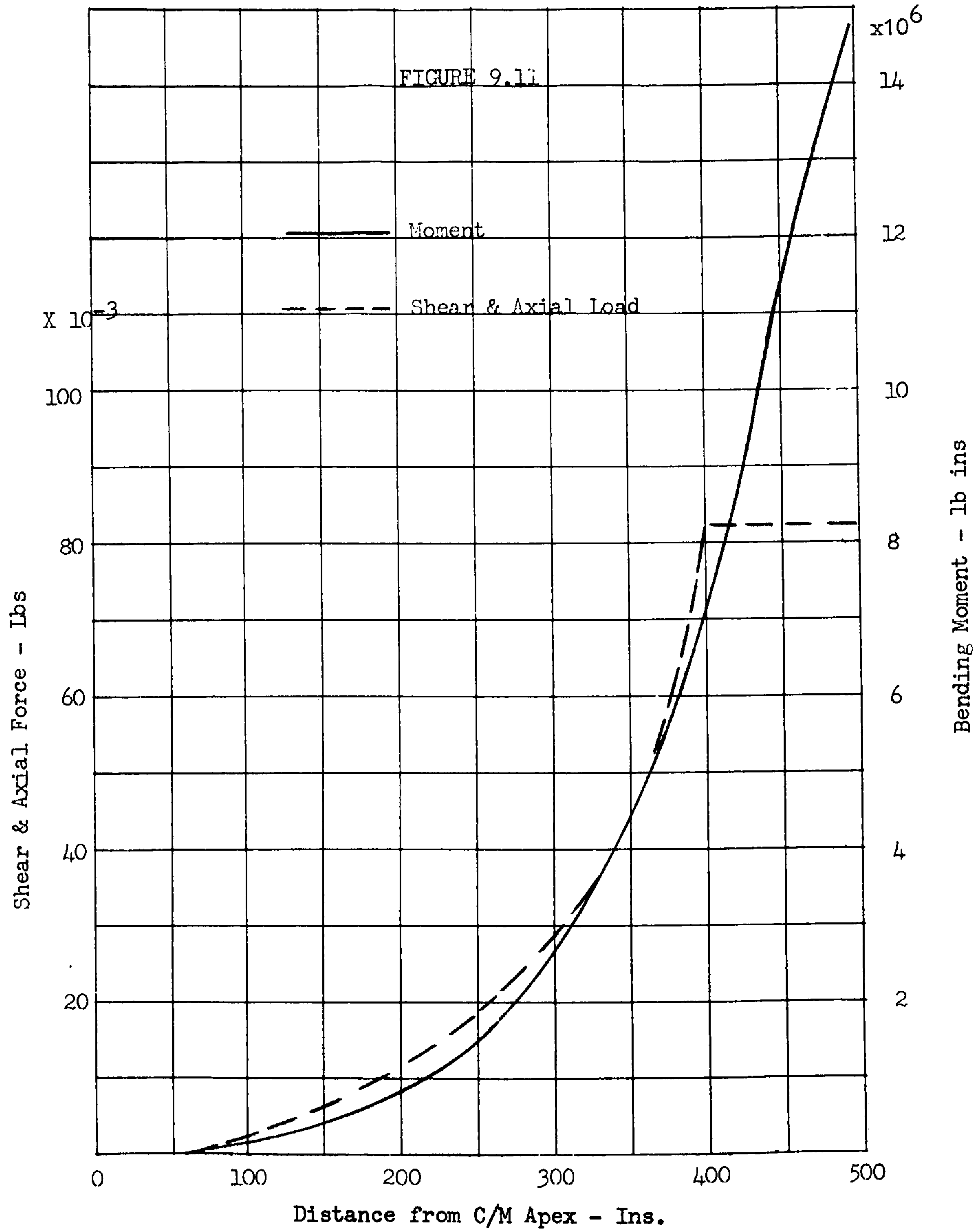

SHEAR & AXIAL FORCE AND BENDING MOMENT

DUE TO UNIT ACCELERATION AT VEHICLE MAXIMUM WEIGHT

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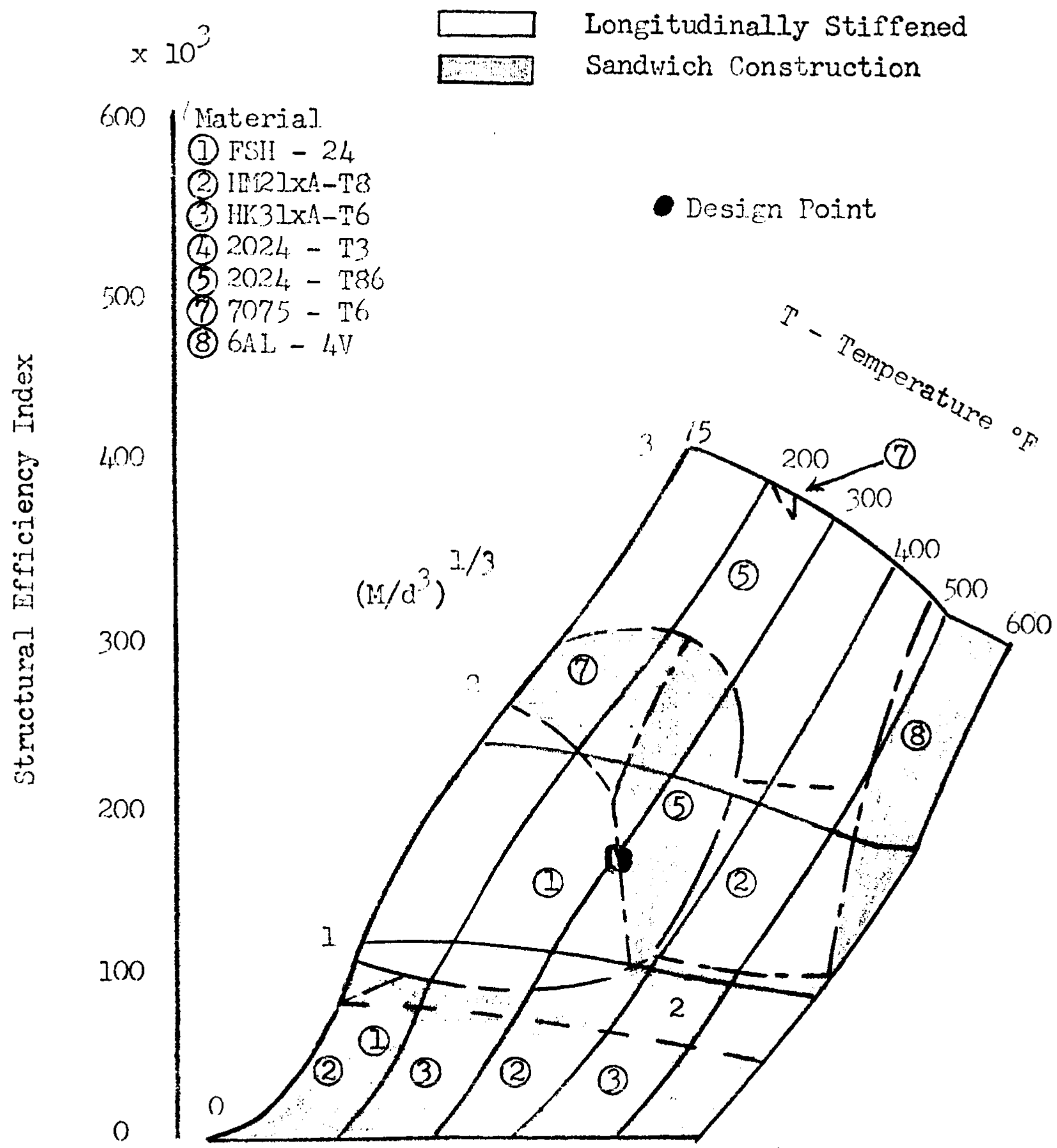
FIGURE 9.12



COMPARISON OF CONSTRUCTION AND

STRUCTURAL EFFICIENCY OF SHELLS IN BENDING the maximum axial compression load. The buckling of honeycomb cylinders under axial compression can be expressed (see Figure 9.13) in terms of the facing sheet thickness, core depths, cylinder radius, and load intensity. Utilizing this figure and accounting for the reduction in allowable stress due to collapsing air-loads, a maximum exterior structure geometry was defined. Optimum core depth also is given in Figure 9.14 for stations 400 to 450.

The propellant tanks, feed lines, and equipment, as mentioned elsewhere, must be protected from penetration by micro-meteoroids. Utilizing the work of Caylor (Reference 3), and assuming the meteoroid flux postulated by Whipple (1961), it was shown that adequate probability of no penetration of the Stage 1 tanks could be accomplished by simply filling the space between the facing sheets of the external structure with a lightweight polyurethane foam. A summary of the penetration characteristics of the vehicle is given in Table 9-IV.

The minimum shell cross-section of the lunar launch stage was determined by investigation of the requirements for meteoroid penetration protection. Reference 3 was the basis of the investigation.

The sections shown in Figure 9.15 are necessary to achieve a probability of .99 that no penetration of the liquid hydrogen tanks will occur during a fourteen-day exposure.

Load calculations based on the data included in Table 9-III were performed to determine the most optimum design section consistent with Figure 9.15. These load intensities were combined with the thermal stresses to evaluate the section requirements. The results were:

1. For the honeycomb construction, the minimum required for

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FIGURE 9.13

REQUIRED SKIN THICKNESS, CYLINDERS IN BENDING FOR VARIOUS LOAD INTENSITIES N_x CORE THICKNESS h_c & CYLINDER RADII (2024 - T86 , T = 300°F)

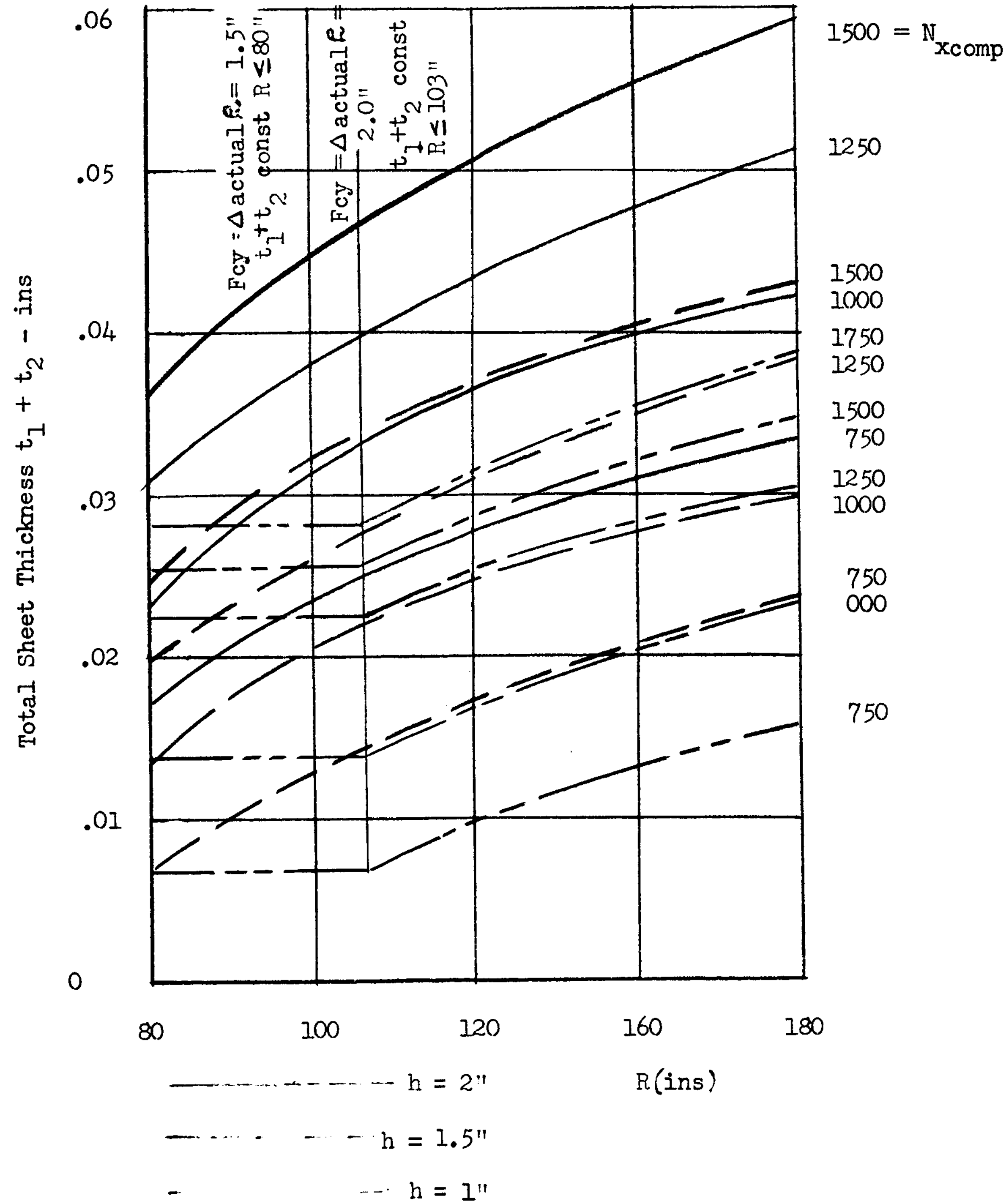

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FIGURE 9.14

OPTIMUM CORE DEPTH - STATION 400-450

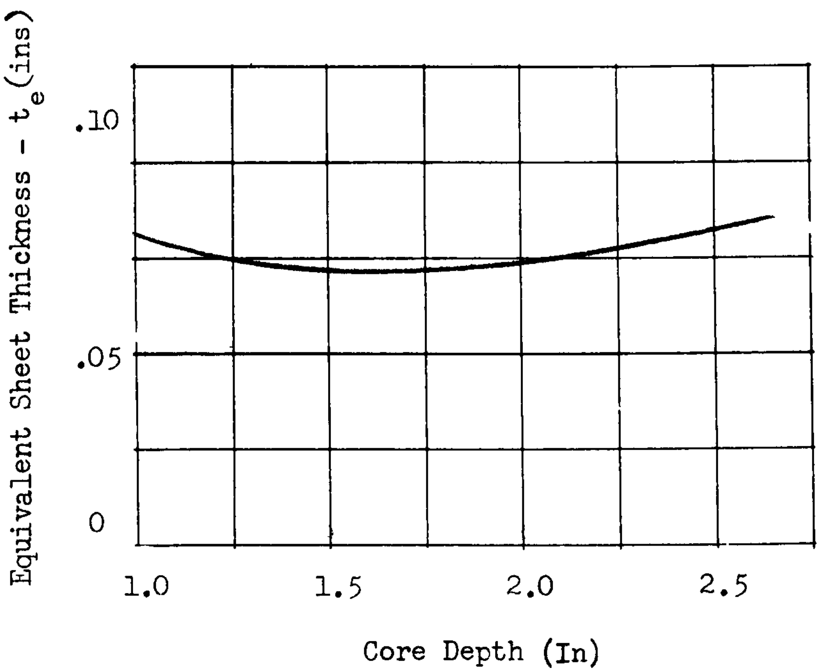

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TABLE 9-IV

PROBABILITY OF NO PENETRATION (P_o) ESTIMATES FOR VEHICLE

Component	Projected Area (Ft^2)	Mission Phase	Time Exposed (T Days)	* $E^{2/3}$	AT (Sq. Ft. Days)	P_o
Stage 1 Tanks	1370	A	3	7.0	4010	0.980
Stage 2 Tanks	600	ABC	14	9	8400	0.995
High Pressure Bottles and Equipment	>10	ABC	14	>10	140	>0.999

Mission Phases:

A = Earth to lunar surface 3 days

B = Extended Moon stay time 8 days

C = Lunar surface to Earth return 3 days

* Estimated from Ref. (3)

[redacted]

FIGURE 9.15

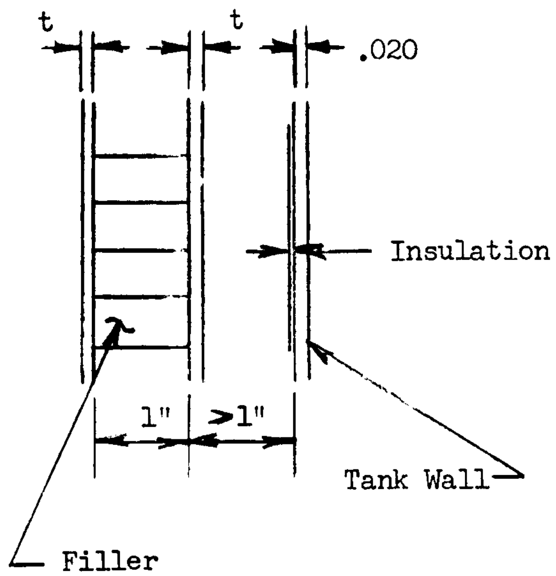
MINIMUM SKIN REQUIREMENTS FOR METEOROID PENETRATION PROTECTION

PARAMETERS – $A = 550 \text{ ft}^2$ $V = 40$ km/sec

$T = 14$ days $P_o = .99$

Material – steel

Honeycomb Sandwich Construction W/Filler



$\underline{E^{2/3} = 4.95}$

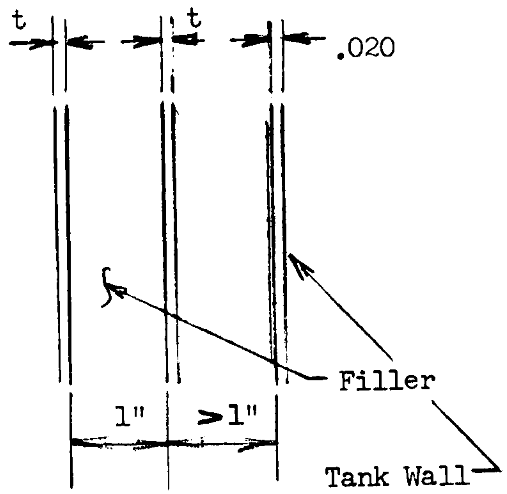
$S_1 = .260$ from Ref. (3)

$\Sigma t_{reqd} = .052$

$t_{min} = \frac{1}{2}\left[.052-020\right]$

$\underline{t_{min} = .016"}$

Double Wall Sheet Stringer Construction W/Filler



$\underline{E^{2/3} = 8.75}$

$S_1 = .260$ from Ref. (3)

$\Sigma t_{reqd} = .030$

$t_{min} = \frac{1}{2}\left[.030-020\right]$

$\underline{t_{min} = .005"}$

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meteoroid protection was more than adquate to sustain the loads imposed and is shown in Figure 9.16.

2. For the two-wall sheet and stringer construction, the minimum secton required for meteoroid protection was also adequate to sustain the loads imposed. However, a face sheet thickness of .005" may be unsuitable in regard to aerodynamic flutter and acoustic vibration. Since the time allotted did not permit a thorough analysis of these effects, a minimum face thickness of .008" was selected on the basis of previous experience in these areas. Prior to final design, an investigation of these effects would be conducted. Stringers to resist the imposed loads were selected on the basis of minimum weight; hence, the minimum section shown in Figure 9.16 was derived.

A comparison of the weight of each construction is shown in Figure 9.16. The double wall sheet stringer construction is lighter.

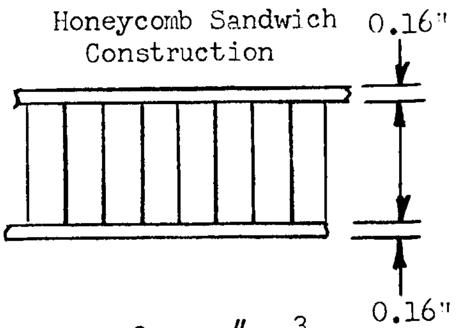
Since the geometric configuration approximates a right circular cone frustrum, the problems of honeycomb construction are numerous, while conventional fabrication techniques can be employed in the double wall sheet stringer design.

In view of the above, the double wall sheet stringer configuration was selected.

9.3 LANDING GEAR DESIGN

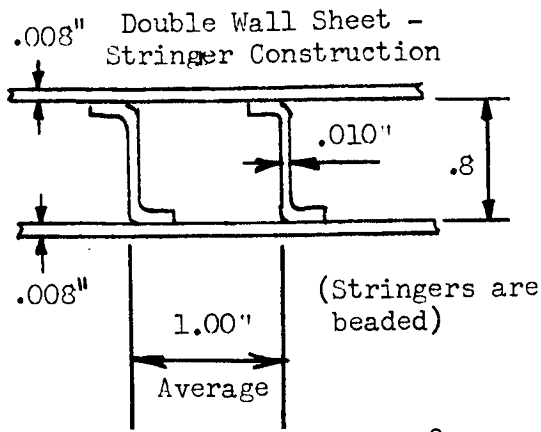
Since the landing gear for the proposed vehicle represents a relatively high percentage of the total vehicle weight, this component was studied at some length. Parameters included in obtaining a landing system weight are as follows:

1. Variable leg spread to center of gravity height ratio.



$\rho_c = 4^{\#}/ft^3$

$\rho_{fibre\ glass} = 2^{\#}/ft^3$



$\rho_{fibre\ glass} = 3^{\#}/ft^3$

ITEM		In/In^2
Face		.0320
Brazing		.0050
Core		.0083
Fibreglass		.0041
TOTAL	=	.0494
Unit Wt.	=	$2.12^{\#}/ft^2$

ITEM		In/In^2
Face		.0160
Brazing		.0019
Stringer		.0150
Fibreglass		.0062
Frames		.0050
TOTAL	=	.0441
Unit Wt.	=	$1.89\#/ft^2$

FIGURE 9.16

WEIGHT COMPARISON OF OPTIMUM

HONEYCOMB AND DOUBLE WALL SHEET -

STRINGER CONSTRUCTION

2. Impact acceleration and/or attenuation stroke length.

3. Landing gear attachment frames.

Following normal optimization procedures, the total landing gear weight and associated frame weights (as a function of vehicle landing weight), illustrated in Figure 9.17, were developed. It will be noted that a weight varying between 3 and 5 percent of the total landing system weight will be necessary, depending upon the landing gear radius and stroke length selected. For minimum landing gear weight, the softest landing system is desirable. This is also consistent with the requirement formulated as a result of landing dynamic studies, which shows that minimum springback (or bounce) should be sought.

9.4 TANK DESIGN

It was necessary to select a tank arrangement which would provide as much tank volume as possible in the space available within the shell. The presence of the rocket engines in the bottom center area, and separation of the take-off stage from the landing stage, also influenced the tank arrangement. In order to store as much propellant as possible within the previously determined shell diameter, toroidal tanks were selected for H_2 and O_2 in the landing stage. The location of the rocket motors within the torus O_2 tank enabled the landing and take-off center of gravity of the vehicle to be lowered considerably, which improved the stability characteristics of the vehicle during landing.

9.4.1 Support Design

Three alternate tank support methods, designated A, B, and C, were analyzed as described below:

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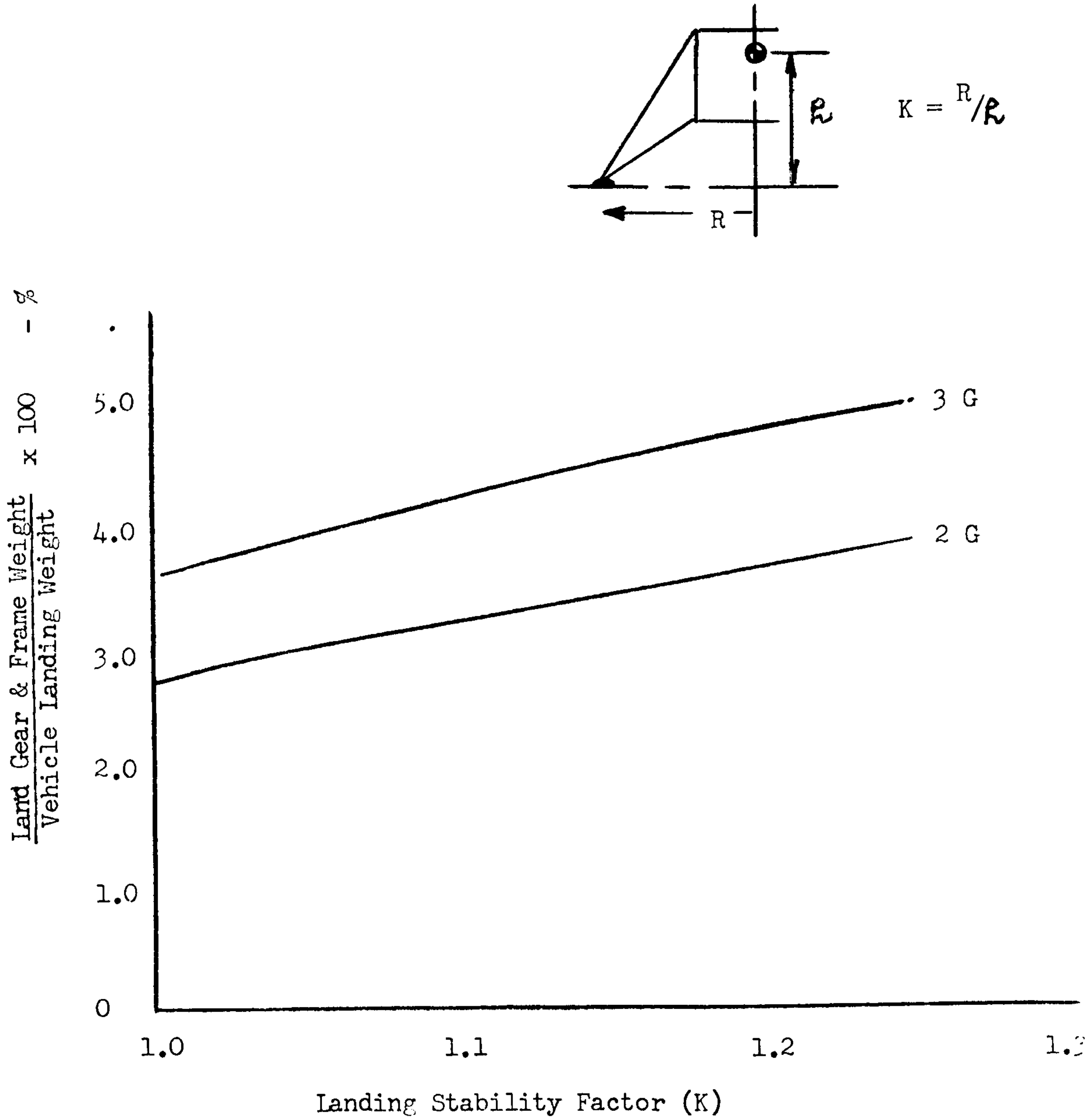

FIGURE 9.17

LANDING GEAR WEIGHT FOR VARIOUS IMPACT LEVELS

AND STABILITY FACTORS

Method A

Support at isolated points equispaced around the large circumference of the toroidal tanks and at points near the outer perimeter of spherical tanks.

Method B

Continuous support around the outer perimeter of toroidal or spherical tanks.

Method C

Continuous support by means of saddle, which in the case of the landing stage tanks is suspended from the outer shell and from the inner cone structure adjacent to the take-off stage compartment.

An analysis was made of the effect of each support method upon the magnitude and direction of the resulting stress distribution in the tank wall and in the adjoining structure, which resists the tank support loads.

The tanks first were analyzed for the maximum and minimum internal pressure conditions and the resulting stress distribution was determined. From this data, a minimum wall thickness for each tank was established. The stress distribution due to the support reactions resulting from each support configuration then was determined and superimposed upon the stress distribution due to the minimum internal pressure condition, since this will be the internal pressure maintained during the boost phase. It was assumed that the tank support loads would only be critical during the boost phase because of the high acceleration loads experienced during this portion of the exit trajectory.

It was found that, especially in the case of toroidal tanks, support methods A and B both caused relatively large compressive stresses in various areas of the tank shell. In addition, method A also caused large shear stresses in the areas near the support frames at the isolated support points. With the exception of the inner cylindrical portion of the first stage H_2 tank, support method C did not produce significant compressive stresses in the tank walls, since the tensile stresses due to the minimum pressure condition were sufficient to cancel any compressive stresses caused by the saddle supports. Therefore, the maximum internal pressure condition dictated the required propellant tank skin thicknesses for support method C.

In the areas of the tank where compressive stresses exist, the tank wall must be stiffened to provide elastic stability. One way to accomplish this would be to increase the skin thickness until the shell possesses sufficient buckling rigidity as a monocoque shell. The gage of the entire outer wall area would have to be increased to the thickness required to provide monocoque buckling rigidity in the areas of maximum compressive stress. A more efficient method would be to add longitudinal and/or tengential stiffening in the local areas of compressive stresses, as required, to provide the necessary buckling rigidity.

A preliminary stress analysis of the tanks, tank supports, and internal structure, together with layouts of required rings and frames, is given in Section 3 of Reference 2, which also gives the procedure employed and detailed results of this analysis. The additional structural members required for each support method is given in terms of either cross-sectional area or as a distributed thickness ($\bar{t}$) over a specified portion of the tank shell. For this summary, the material volumes and distributed thicknesses required for each tank and support method is reduced to an overall $\bar{t}_{ave}$

value for each tank, which includes the basic skin thickness dictated by the maximum pressure condition. A t_{ave} value is also given, which is the difference between the actual monocoque thickness required for maximum pressure and the average distributed material thickness (t_{ave}). Values for t_{ave} and t_{ave} are given in Table 9-V, which also includes the weights of the major structural components and total tank weight for each support method.

REFERENCES

1. Report SID 62-1189, "Lunar Mission System Studies - Interim Report," dated October 1962

2. J. Sandford, P. T. Mikkelson, C. D. Hill, L. G. Drankin, Summary Weight and Structure Analysis of the Direct Mission Vehicle, NAA-S&ID, September 1962

3. G. H. Caylor, Preliminary Survey of Meteoroid Effects on Space Vehicles, NAA, SID 62-519, April 1962

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Table 9-V. Values for Average Distribution Thicknesses of Skins, Stringers, and Frames of Landing Stage O_2 and H_2 Tanks for Alternate Support Methods.

1	2	3	4		5		6		7		8	
TANK	Support Method	Number of Isolated Supports ("A" only)	Average Distributed Material Thickness and Weights of Major Structural Components						Basic Average Skin Thickness and Tank Weight Required for Maximum Pressure Conditioning		Difference Between Total Distributed Thickness and Basic Thickness (6 - 7)	
			Skin, Stringers, and Intermediate Frames		Major Frames		Total					
			$\bar{t}_{ave}$ (in)	Weight (lb)	$\bar{t}_{ave}$ (in)	Weight (lb)	$\bar{t}_{ave}$ (in)	Weight (lb)	$\bar{t}$ (in)	Weight (lb)	Δt (in)	ΔW (lb)
O_2	A	3	0.038	363	0.0105	100	0.049	463	0.032	305	0.017	158
	A	4	0.037	347	0.0113	107	0.048	454			0.016	149
	A	6	0.033	312	0.0131	124	0.046	436			0.014	131
	B	-	0.069	659	0	-	0.069	659			0.037	354
O_2	C	-	0.032	305	0	-	0.032	305	0.032	305	0	0
H_2	A	3	0.108	1942	0.019	347	0.127	2289	0.061	1093	0.066	1196
	A	4	0.103	1856	0.018	322	0.121	2178			0.060	1085
	A	6	0.090	1621	0.016	287	0.106	1908			0.045	815
	B	-	0.089	1599	0.014	260	0.103	1859			0.042	766
H_2	C	-	0.061	1093	-	-	0.061	1093	0.061	1093	0	0

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10.0 WEIGHTS

In order to accomplish a direct Apollo lunar landing, the Earth escape weight cannot exceed 90,000 pounds. With the use of cryogenic propellants in the lunar landing and take-off stages and by reducing the weight of the Apollo payload to 10,000 pounds, an escape weight of 86,320 pounds can be achieved. Table 10-I is a weight breakdown of the Apollo payload and Table 10-II is a weight breakdown of the direct mission vehicle.

A description of the weight savings can be found in Lunar Mission System Studies Interim Report SID 62-1109, dated 30 October 1962.

TABLE 10-I

APOLLO PAYLOAD WEIGHT BREAKDOWN

COMMAND MODULE	(7,323)
Structure	3,333
Crew Systems	1,305
Communications and Instrumentation	760
Guidance and Navigation	250
Stability and Control	135
Reaction Control	355
Electrical Power	356
Environmental Control	424
Earth Landing	405
SERVICE MODULE EQUIPMENT	(2,573)
Electronics	139
Reaction Control	899
Electrical Power	983
Environmental Control	552
CONTINGENCY	(104)
TOTAL PAYLOAD	10,000

TABLE 10-II

PRELIMINARY WEIGHT STATEMENT

LIM/SM-APOLLO DIRECT MISSION

Stage II		Pounds
Structural Section		(1,836)
Equipment Provisions		75
Structural Interstage		1,171
Inner Portion		335
Fiberglass		150
Polyurethane Foam		90
Doublers - Attach S/M to C/M		15
Propulsion System		(1,307)
Propellant Tanks, Insulation, Support		597
LO_2 Tank	267	
LH_2 Tank	330	
Pressurization System		238
Propellant Feed System		32
Engine Installation		440
Trapped Propellant		246
Residual Vapor		80
Booster Burn-out Weight		3,469
Propellant		14,620
LO_2	12,185	
LH_2	2,435	
Gross Weight, Booster Stage II		18,089
Payload		
Command Module		6,825
Service Module		3,175
Material Left on Lunar Surface		- 170
Propellant Boil-off		146
Lunar Launch Weight		28,065

CO[redacted]

Stage I

		Pounds
Structural Section		(2,415)
Outer		1,550
Inner		490
Polyurethane Foam		375
Propulsion System		(4,243)
Propellant Tanks, Insulation, Support		2,430
LO_2 Tank	705	
LH_2 Tank	1,725	
Pressurization System		788
Propellant Feed System		117
Engine Installation		908
Landing Gear		1,390
Trapped Propellant		882
Residual Vapor		240
Material Left on Surface from C/M		170
Lunar Landing Weight		37,405
Propellant		48,285
LO_2	40,240	
LH_2	8,045	
Propellant Boil-off		630
Escape Weight		86,320